



NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
MARSHALL SPACE FLIGHT CENTER, ALABAMA 35812

REPLY TO
ATTN OF: PD-TUG-C-73-189

August 24, 1973

TO Addressees

FROM PD-TUG-C/Mr. Orillion

SUBJECT Tug Mission Extension - Task 6

The attached report is the completed effort by MDAC under the special emphasis of Task 6. These data will be used and extrapolated to meet the requirements of the servicing mission of Option 2.

Alfred G. Orillion
Alfred G. Orillion

1 Enc:
as stated

Addressees:

- PD-TUG-MGR, Mr. Teir/Mr. Huber
- PD-TUG-MGR, Mr. Weir
- PD-PL, Maj. Feibelman
- PD-TUG-M, Mr. Rutland
- PD-TUG-P, Mr. Stucker
- PD-TUG-E, Mr. Sisson/Mr. Laue
- PD-TUG-E, Mr. Recio
- PD-TUG-E, Mr. Saucier
- PD-TUG-C, Mr. Barrett
- PD-DO-DIR, Mr. Gregg
- PD-DO-E, Mr. Sanders
- NASA Hq/MTE, Mr. Schnyer
- Aerospace, Mr. Schilb
- SAMSO(XRZ), Capt. Probst

(NASA-CR-179104) SPACE TUG SYSTEMS STUDY
(CRYOGENIC): SPECIAL EMPHASIS MISSION
EXTENSIONS Draft Report (McDonnell-Douglas
Astronautics Co.) 106 p Avail: NTIS

N87-70402

Unclas
00/18 0074240



SPACE TUG SYSTEMS STUDY (CRYOGENIC)

DRAFT REPORT

TASK 6 - SPECIAL EMPHASIS
MISSION EXTENSIONS

AUGUST 1973

PREPARED FOR NATIONAL AERONAUTICS AND SPACE ADMINISTRATION,
MARSHALL SPACE FLIGHT CENTER,
UNDER CONTRACT NO. NAS8-29677

MCDONNELL DOUGLAS ASTRONAUTICS COMPANY-WEST

5301 Bolsa Avenue, Huntington Beach, CA 92647

1.0 INTRODUCTION AND SUMMARY

During the course of the main tasks of the cryogenic space tug study missions in excess of 6 days duration were not considered. This was the result of no missions in the DOD-NASA mission model requiring orbit stay times in exceeding this period. However, it is possible that in future times missions will become of interest where the tug is required for periods of longer duration. Therefore, within the context of the special emphasis task (Task 6) an assessment of mission extensions was made.

The next sections of this report present the results of a study of mission extension possibilities. Section 2 covers a description of the study with discussions of the groundrules, guidelines and assumptions made. Section 3 presents the technical details of the analysis. The last section provides a brief recommendation for future activities along the lines of mission extensions if this subject should be pursued.

In summary, the results of the study show that the Cryogenic Space Tug offers significant potential for extended mission durations out to 6 months. The primary effects that limit increased stay times are propellant heating and boiloff as well as extra consumables for increased watt-hours of electrical power. However, by modest increases in the multi-layer insulation for the cryogenic tanks the boiloff can be reduced markedly. By means of a scheme described in detail in Section 3 the oxygen losses due to venting can be eliminated, thereby decreasing the boiloff by a factor of 2.

At this first level of analysis system reliability can be projected for mission extensions only statistically. Lacking detailed data on service life of components so that wearout could be considered, an intuitive approach was taken to the effects of increased operating time failures. The study inferred that other factors and considerations such as ground handling environments, maintenance and servicing procedures, orbital deployment and recovery, and so forth might result in a very reliable tug system for the tug mission segment of the overall tug-shuttle ground and space regime. The subject of course requires further study when more definitive data become available. However, the calculations show that the mission completion reliability estimate for the tug systems exceeds 92 percent for a 30-day synchronous orbit mission.

2.0 STUDY DESCRIPTION

The objective of the study was to include an evaluation of the effects on the Tug systems to increase the in-space stay time beyond six days. An assessment was to be made to determine as stay time is increased what thresholds would be encountered that would require major subsystem changes to operate at (1) the same continuous level, (2) various reduced levels and (3) intermittently. For each mode of operation changes to the Tug systems and/or options that could be imposed to meet different time limits were to be identified. Within the limits and resources available for these tasks, these objectives have been met.

Extended missions examined by the study were largely hypothetical with the specific exception of two service missions to four satellites in synchronous equatorial orbit. For these two missions, timelines were provided to the study team describing the events covering total periods of thirteen and thirty days duration respectively. This class of mission was examined in the first order of detail parametrically regarding the propellants consumed as a function of geosynchronous stay time and are described elsewhere in this report. It is sufficient to state at this time that the study concluded that the 6-day direct developed tug is capable of servicing four satellites in the context of the reference missions provided.

Other extended missions examined included lunar missions and libration point visits which might prove of future interests to scientific exploration programs. Within the category of low earth orbits loiter missions and long duration payload attendances mission which might be desirable for achieving shuttle flexibility and military consequences were also included. The Table 2-1 summarizes the classes of missions examined.

TABLE 2-1
EXTENDED STAY MISSION POSSIBILITIES

MISSION TYPE	DURATION (DAYS)	DELTA V (FT/SEC)	ROUNDTRIP PAYLOAD (LBM)	DEPLOYMENT PAYLOAD (LBM)
GEOSYNCHRONOUS	13-30	29,000	3,300	8,600
LUNAR	14-28	28,000	3,900	9,900
LIBRATION POINT L ₁	12-18	24,000	6,500	14,700
LIBRATION POINT L ₂	18-24	23,000	7,800	16,800
PRE-MISSION LOITER	60 MAX	28,500*	3,000	7,700
PAYLOAD ATTENDANCE LOITER	180 MAX	27,000*	3,000	7,300

* DELTA V CAPABILITY (WITH 3,000 LBM PAYLOAD) REMAINING AFTER MAXIMUM DURATION
LOITER IN 160 N.M. ORBIT

At the outset of the study it was necessary to define the Tug system to be assessed for extended stay times and to establish groundrules to form the framework of the analysis. Realizing that specific recommended configurations would not emerge from the mainline study activities in sufficient time to proceed with the work of Task 6, a ground rule was set defining the Tug to be that which would be responsible to the most severe mission requirements of the overall DOD-NASA mission model. Therefore, a full direct capability option with rendezvous and docking configuration which could satisfy the requirements of the so-called Reference Mission 'Alpha' was defined as baseline. It was further assumed that this configuration would be the most advanced from the standpoint of capabilities including a level II navigation autonomy. Typically configuration options 402 or 403 would be representative of the baseline. For reference purposes pertinent descriptions of configuration numbers 402 and 403 are repeated. Figure 2-1 is a pictorial of the 402 option:

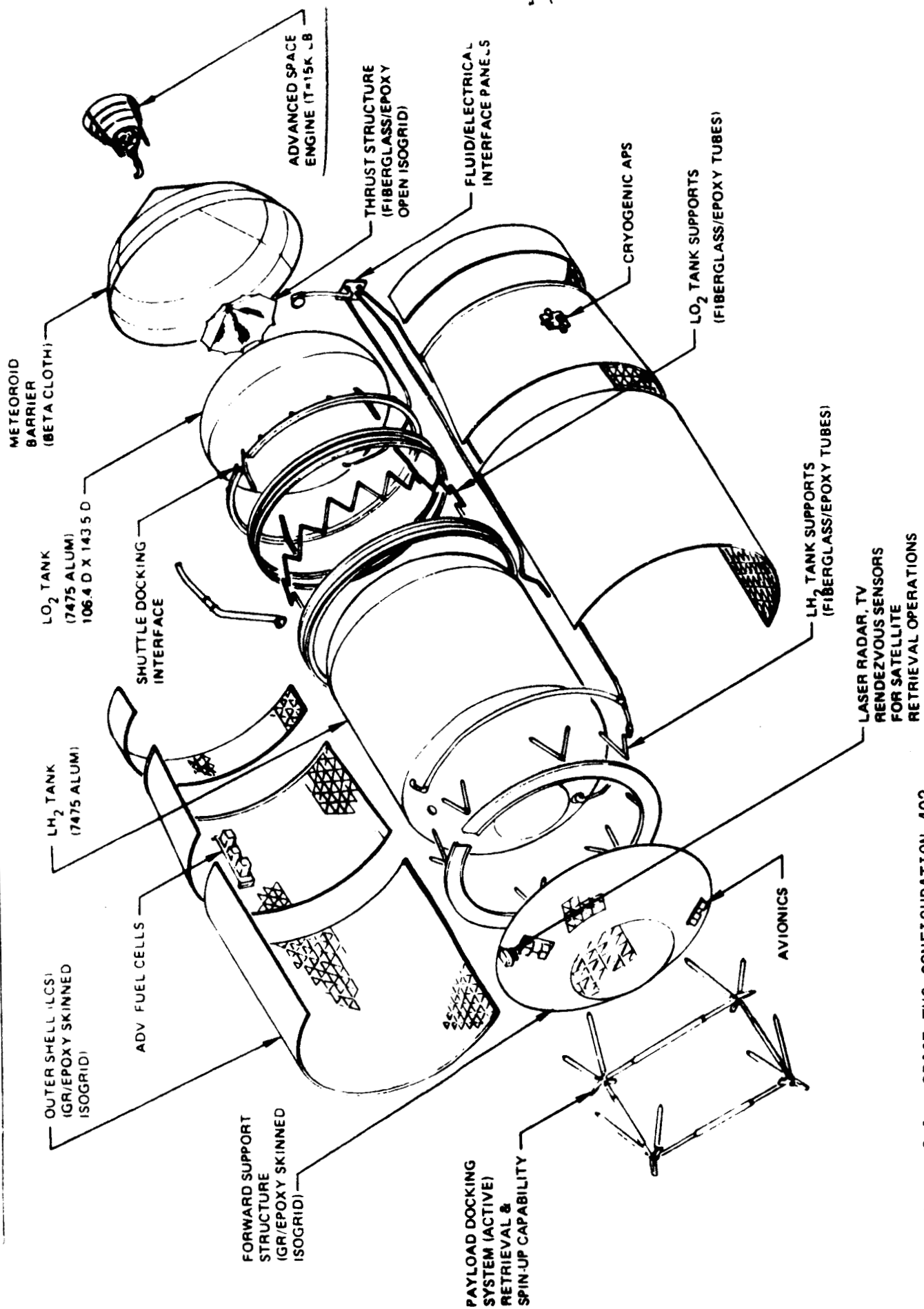


Figure 2-1. SPACE TUG CONFIGURATION 402

Characteristics of space tug considered for mission extensions.

Configuration	402	403
Mission Duration (Days)	6	6
Autonomy Level	II	II
Main Engine	Advanced Space Engine	Aerospike
Mixture Ratio	6:1	5:1
Isp	470	468
Thrust	15 Klbf	15 Klbf
APS	Cryo	Cryo
Thermal Insulation	MLI	MLI
Electrical Power System	Advanced Fuel Cells	Advanced Fuel Cells
Payload	9350 Lb	8990 Lb
to	6060 Lb	5820 Lb
Sync Orbit	3660 Lb	3520 Lb
Deployed Retrieved Round Trip		
Tank Capacity	55.9 Klb	55.8 Klb

Another assumption that was made in the study was that the Tug system service life requirement will allow 20 six-day missions to be accommodated within an optimum turnaround period and with an overall reliability of 0.97 per mission. Maintainability design considerations will allow for manual or automatic replacement of faulty component prior to launch.

The study realized that a more specific, complete and detailed description of the Tug configuration was not available at the outset of the study. In particular, data was lacking on individual components operational and service life characteristics. It was necessary therefore to rely on experience gained in previous studies, current designs and operational experience (including Skylab and Saturn/Apollo) to together with judgment define characteristics and specifics of importance but not available to the study team. These reliances included forecasts in the form of generalized scenarios on how the Tug, within the total context of the operational Shuttle/Tug/payload complex, might be used.

At this point in the study it was realized that the Tug system would be subjected to stresses and environments above and beyond those encountered during the Tug mission segment of its service experience. Forecasts and estimates were made in order to define the periods of high stress to be serviced that would result in design decisions and component selection. The result of these assessments turned up the conclusion that ground handling, launch preparation, shuttle powered flight and recovery operations presented more severe thermal, acoustical, vibration, shock and

acceleration environments than the Tug powered and unpowered flight environments. Further discussion of these aspects of the study will follow.

The study examined several factors which can and do limit mission stay time. These factors can conveniently be divided into two classes: (1) uses and losses of consumables and (2) equipment service life limitations.

The consumables under study include primary propellant oxidizers and fuels, pressurants, attitude control engine propellants, fuel cell reactants and other expendables. The number of cases which require study is considerably simplified by considering the 402/403 class Tug which use the same fuel and oxidizer for all sources of propulsive and electrical power.

Here the only losses that would be significant are boiloff venting of the main tanks due to propellant heating. Leaks in other tanks, feed lines and fluid components can be considered to be negligible. This assumption justifiably is made contemplating that safety considerations will dictate that all lines and valves containing propellants will be as leak free as possible using such techniques as welding and brazing as opposed to made up fittings and joints. When these assumptions are made, the study analyses then can be directed to propellants required for maneuvers and attitude control during the mission and boiloff losses that occur with time.

As for equipment service life limitations, individual components which are subject to wearout, fatigue and degradation with operating time were categorically examined. The study after a perusal of a projected ground/flight mission timeline assumed that all mechanical components with the

exception of the engines and thrusters, would be stressed at lower levels during the Tug mission than during the ground and Shuttle portions of the flight. For example, during Shuttle boost and entry acceleration levels exceeding 3 g's are encountered. Therefore, for these components fatigue failures induced by vibration and shock would not be expected since design limits would be established by the ground and Shuttle environments.

Fluid handling items such as valves and actuators comprise a class of components where service life is in general cycle rather than operating time limited. Therefore, in order to establish a high degree of confidence as to service limits, these components are subjected to extensive life testing procedures during hardware development, production sampling, and qualification tests.

Operational factors favor designs where fluid components which do not require frequent replacement or servicing. Extensive testing, checkout and inspection are procedures required to recertify a vehicle for flight having experienced rework of a faulty valve, line or tank. A design would be favorable where these items would demonstrate a service life in excess of the system service life requirements of an individual Tug. Further examination of a typical extended mission (for example a 30-day synchronous orbit service mission) compared to the six-day "alpha" mission indicates a modest (20% increase) fraction of additional cycles required of the fluid items.

As for the electrical power generation and conditioning components concerned, the limiting factors for the 402 and 403 configuration will be the availability of GOX/GH_2 to provide the reactants for the fuel cells. Fuel cells do experience a slow degradation with operating time as inert substance build up at the electrodes. It is possible and desirable to remove these contaminants by a simple purging operation. This involves the venting of the low pressure side of the cell allowing an excess of gas to sweep foreign substances clear of the cell without the interruption of power being generated. In the process the cell is restored to a fresh state at only the expense of the purge gases. This procedure will not be unique to mission extensions. Purging will occur several times during a normal 6-day mission.

Information from fuel cell suppliers indicates that there is no noticeable deterioration over 5,000 hours of operation, other than what is caused by buildup of inerts prior to the purging described above. The study concludes that no service limitations or thresholds will be encountered during extensions other than the availability of reactants.

The electronic, electromechanical and electro-optical components perhaps deserve scrutiny. In these systems there are a much larger number of individual piece parts all of which need to function together. Some individual failures can be tolerated when redundant elements are included in the design. It can be expected that most of the active elements will be solid state devices, the character of which is to remain functional so

long as the environment suitable for reliable operation is maintained. Mission extension in the first analysis will not adversely alter this favored environment.

One of the major causes of avionic electronic failures is thermal cycling. Since the Tug is being designed to accomplish 20 missions, a stable thermal control system already appears to be a requirement, and therefore the effect of an extended mission should be minimized from the standpoint. The question of the benefits from powered down operation cannot be fully evaluated lacking discrete details on the thermal and electrical environments. However, placing equipment in a standby mode may prove attractive from a reliability standpoint.

Avionics components subject to wearout and fatigue include all rotary devices such as gyroscopes, gimbals, drive motors, to mention a few. These devices, similar to fluid components, can be subjected to extensive life testing in order to determine service life limits. It has been observed in prior satellite programs that the orbital null-gravity environment is far less severe than the terrestrial environment experienced during ground testing and checkout from the standpoint of shock and vibration as well as gravity induced loads. Therefore, it can be safely assumed that adequate test data will be available to determine what rotating components would not be likely to survive an increase in orbit stay time. On the other hand if such components were to exist then they would be subject to rework, replacement and servicing after and prior to each mission. These components

would be the weakest link in the chain so to speak and if there service life could be substantially extended by appropriate design changes, those changes would be highly desirable. Along this line of thought, the study concludes that the service life of rotary components will be adequate for extended missions.

Another class of avionic components subject to wearout is thermionic devices such as travelling wave tubes, flash tubes, magnetrons, klystrons and similar vacuum tube type devices. However, it must be remembered that the state-of-the-art is continually advancing as newly developed techniques emerge where thermionic devices are replaced by solid state components of comparable or vastly superior performance, including substantial increases in lifetime due to lower power dissipation and operating temperatures. Optical components such as mirrors, prisms, lenses, gratting, and filters are passive devices and as such not subject to fatigue failures. However, some degradation with time is experienced as the critical surfaces become contaminated when exposed to the environment surrounding the orbiting vehicle. However, time dependent effects of effluent impingement on the degradation of optical surfaces and components for the purpose of this analysis are difficult quantifying. Some limited data and experience is available from prior manned and unmanned spacecraft.

A synopsis of the general approach that the study applied to assessment of mission extensions from an equipment functioning point can be articulated.

Until the design has advanced to the point where numerical data is available or failure thresholds for wearout and fatigue components it is sufficient to assume that the exhaustion of consumables is the first threshold to be experienced. When life test data becomes available, higher order assessments of individual wear items can be ascertained.

The technical thrust of the study proceeded along two principal lines of analysis. ^① The first consisted of considerations of the vehicle flight characteristics and orbital mechanics in terms of propellants required for such functions as rendezvous maneuvers, stationkeeping and control functions as mission time progressed. ^② The second technical factor assessed was that of propellant heating in terms of cryogenic losses as a function of mission time. In this assessment optional values in terms of multi-layer insulation weight were computed for various mission periods. Because propellant weight consumed or lost and mission time were common factors to each analysis area, the two results could be compared along common grounds. What these analyses show are that the rate of consumption of propellant for extended missions is small in the context of the overall Tug mass budget.

^③ A more specific analysis was made of the requirements for meteoroid protection. In this area it was observed from the results of the study that adequate protection can be expected for stay time extension. This conclusion is reached when comparing the modest requirements for meteoroid protection to the much larger requirements for thermal considerations.

④

Further limited assessments were made in the electrical power and systems reliability areas. The data, from these as well as the other individual analyses, are found in the following section.

At the outset of the study attention was given to the possible advantages of using slush or triple-point hydrogen, in place of the cryogenic liquid, to mission extensions. The effort was of rather limited scope and drew upon prior experience to a large degree. Slush hydrogen is of considerable interest as a propellant when tank volumetric limitations are encountered because of its high density and heat capacity over the liquid. These characteristics allow combinations of lower tank weight and size, longer storage in space and the possibility for lower insulation weight. Off-setting disadvantages include ground handling complications in making slush out of the saturated liquid, tug changes to include baffles and screens as well as agitators and stirring devices to prevent clogging and promote flow of fuel to the engine. Appendix A contains information on the characteristics of slush hydrogen. Calculations reveal that a 50% slush mixture will sustain an orbital stay period of over 60 days duration without appreciable loss of hydrogen due to boiloff.

3. TECHNICAL ANALYSIS AND RESULTS

Possible extended missions were summarized in Table 2-1. Also included in this table are estimates of velocity requirements and payload weights, assuming both roundtrip and deployment missions. These missions were used to examine impacts on propellant consumed, system operation and reliability forecasts for extended stays.

3.1 FLIGHT MECHANICS AND ORBITAL DYNAMICS CONSIDERATIONS

A series of analytic effects were attempted in order to examine and assess the impact of mission extensions on tug energy requirements. In this section, the results of fixed analyses are present along two general lines. The geosynchronous missions where several satellites are to be visited, are examined in order to determine the energy requirement for orbital maneuvers. The loiter missions on the other hand are examined from the standpoint of stationkeeping and attitude hold energy requirements. The lunar and libration point missions are treated lastly in more general terms.

Equatorial Synchronous Orbit Missions

Arbitrary Longitude Transfers

Figures 3-1 through 3-3 form a package which allows the user to evaluate ΔV and propellant requirements as a function of time for an arbitrary longitude transfer in equatorial synchronous orbit (ES0). The transfer is performed by:

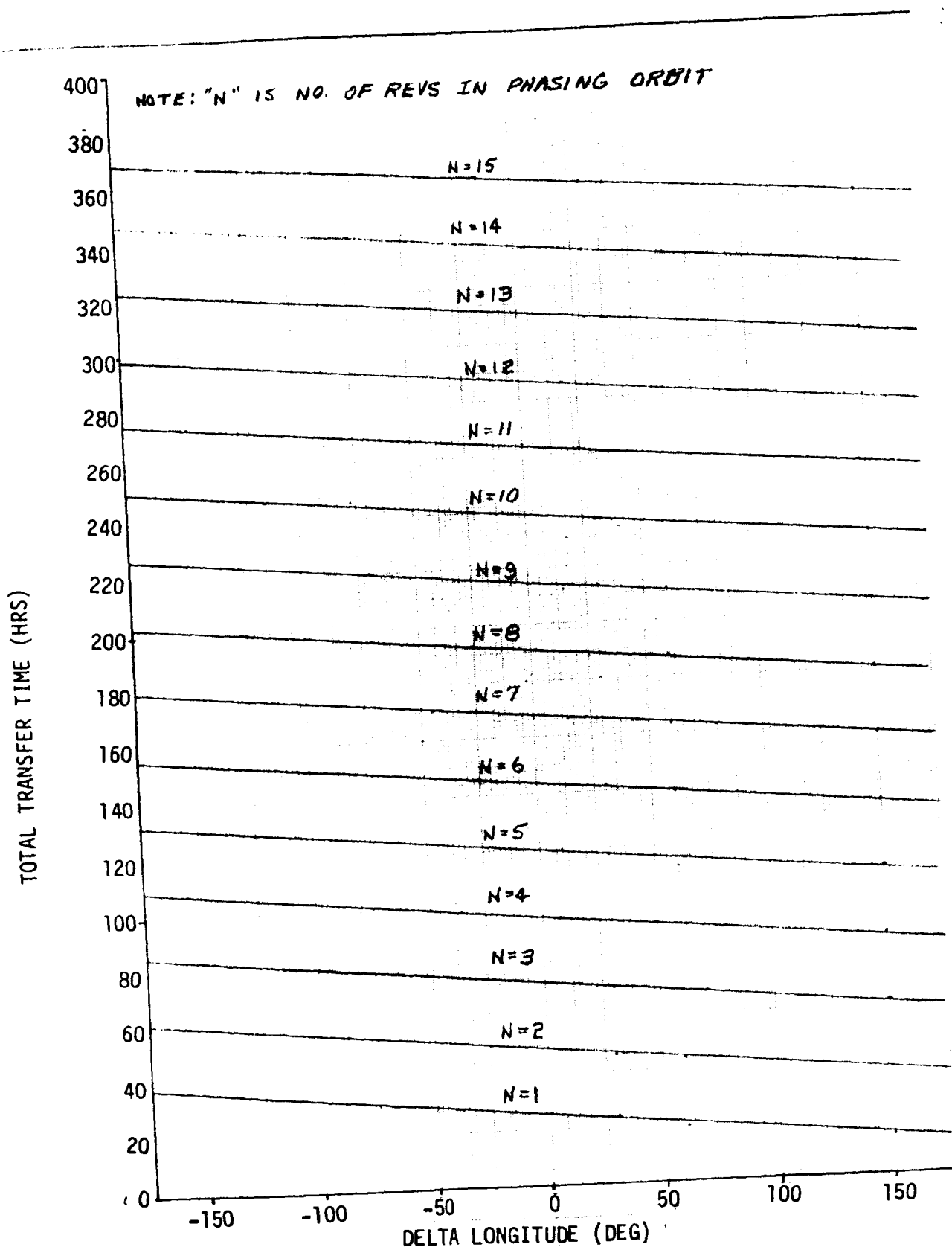


Figure 3-1. TIME REQUIRED FOR LONGITUDE CHANGE IN EQUATORIAL SYNCHRONOUS ORBIT

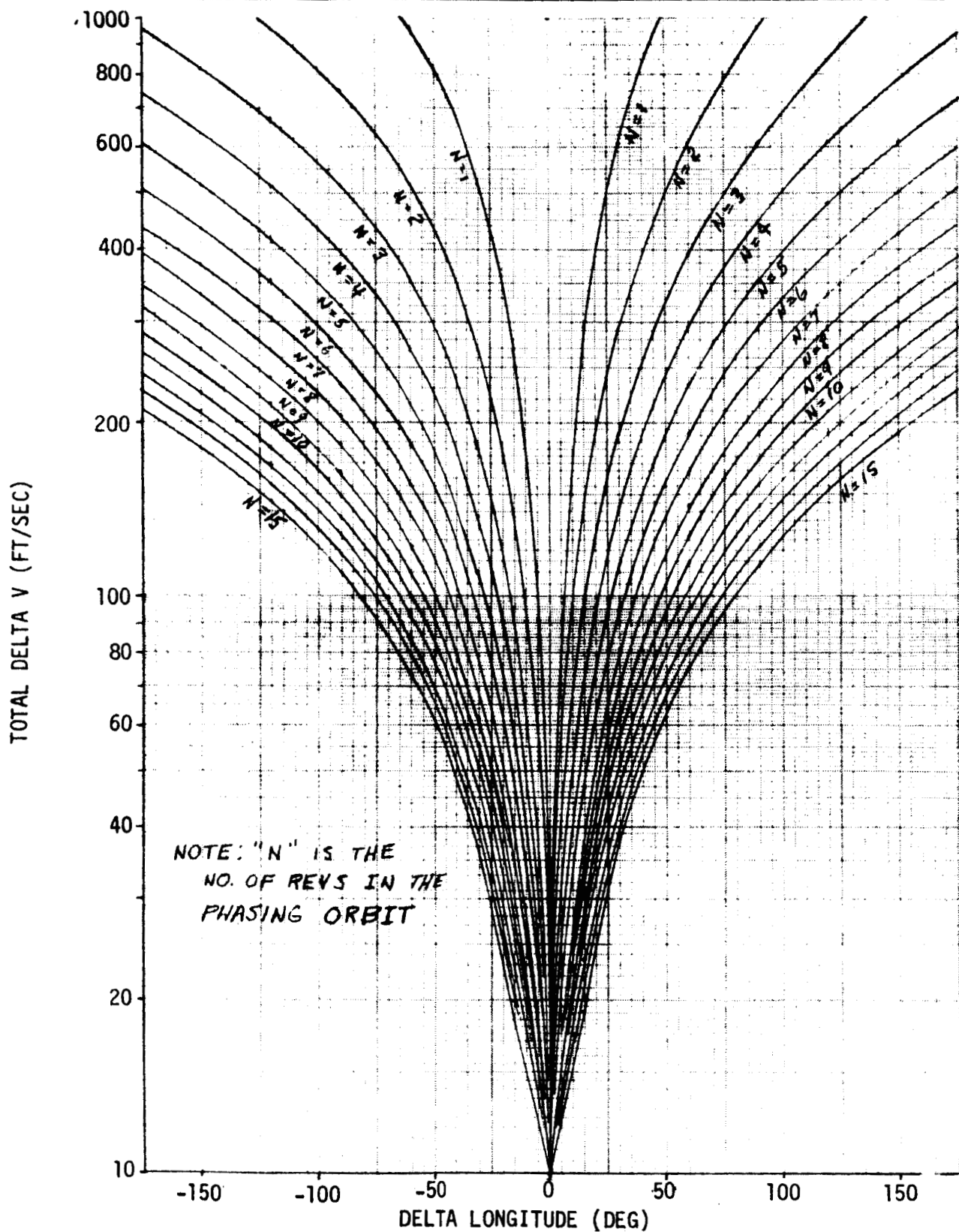


Figure 3-2. VELOCITY REQUIREMENT FOR LONGITUDE CHANGE IN EQUATORIAL SYNCHRONOUS ORBIT

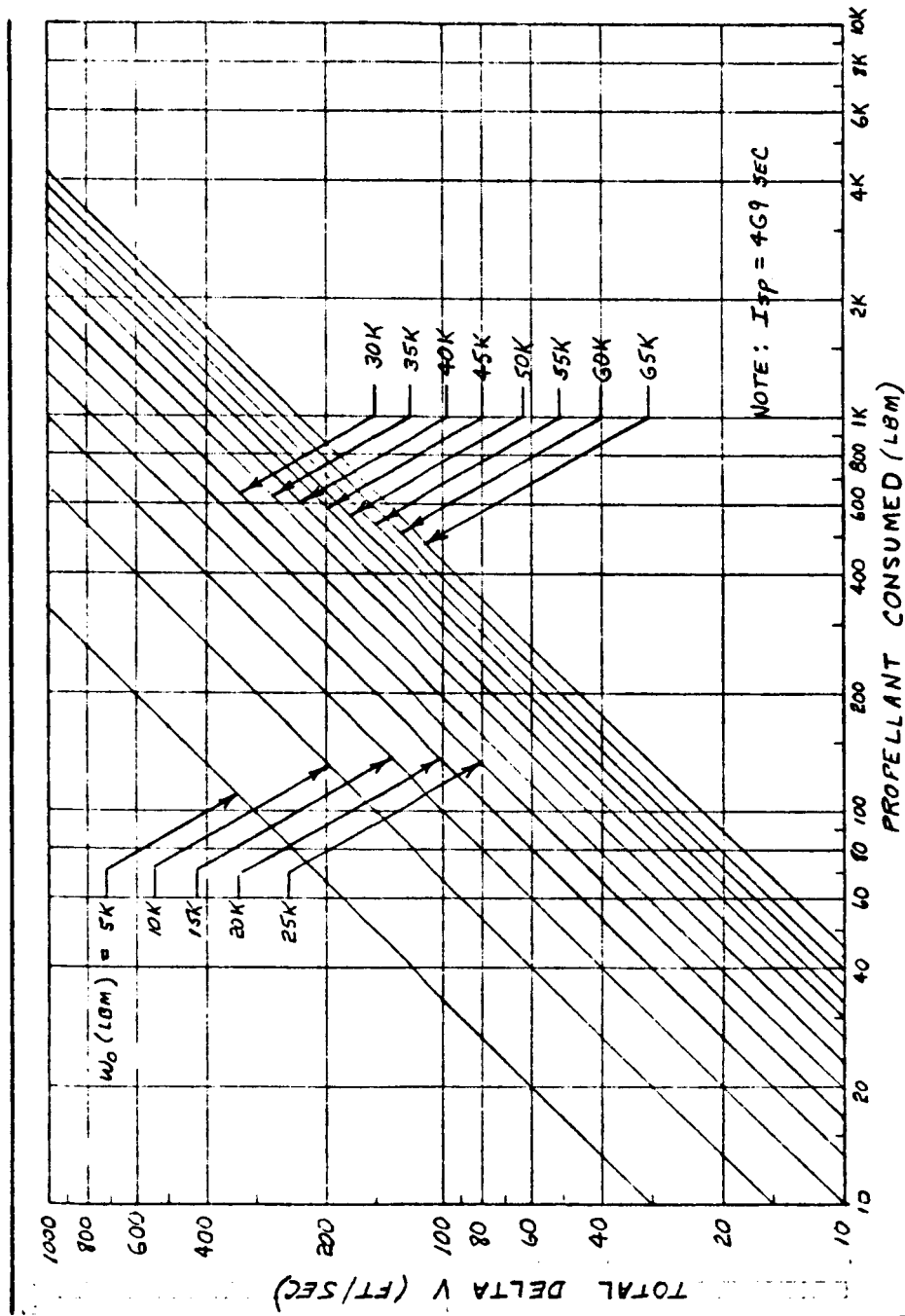


Figure 3-3. PROPELLANT REQUIREMENTS

- (1) Applying an impulse (ΔV_1) to go into a phasing orbit
- (2) Coasting in the phasing orbit for an integer number of revolutions (N)
- (3) Applying an impulse (ΔV_2) equal and opposite to ΔV_1 to go back to EOS

For transfers where the desired change in longitude is positive the phasing orbit will have a perigee less than synchronous altitude, with a shorter period. For negative transfers, the phasing orbit altitude will be higher than synchronous altitude with a corresponding longer period. This effect can be seen in Figure 3-1 where total transfer time is presented as a function of delta longitude with the number of revolutions in the phasing orbit parameterized from 1 to 15. For a given number of revolutions in the phasing orbit, the total transfer time, which is just "N" multiplied by the period of the phasing orbit, decreases with increasing delta longitude.

Knowing the time available for the transfer a value of "N" can be chosen from Figure 3-1. Using that "N" a value of ΔV can be obtained from Figure 3-2. The ΔV plotted in Figure 3-2 is the total velocity required to go into and out of the phasing orbit ($\Delta V_1 + \Delta V_2$). Note that it is slightly more efficient to transfer backward than forward. If the initial weight (W_0) of the Tug is known at the beginning of the maneuver, Figure 3-3 will provide a conversion from ΔV to propellant consumed.

Baseline ESO Service Mission

Although Figures 3-1 through 3-3 are sufficient to analyze a single transfer in ESO, when multiple transfers are required during the same mission additional factors come into consideration. What is the most efficient way to split the total available time between the several transfers which have to be performed? The number of possible ways to split the total time between the transfers goes up rapidly with both the number of transfers and the total time available. It remains finite, however, since there is always an integer number of revolutions in each phasing orbit. This fact allows the problem to be reformulated. Given a total number of revolutions in phasing orbit, what is the best way to split them between the several transfers. A small program has been written which performs a hunt on the possible phasing orbit splits and finds the one which minimizes the total ΔV required.

This program was used to analyze the baseline ESO service mission. The mission involves servicing four satellites in ESO which are at longitudes 250° , $285^\circ - 75^\circ$, and $115^\circ F$ respectively. Thus three transfers of 35° , 150° , and 40° are required. Table 3-1 presents the effects of time available on the mission profile. The times presented in this table refer to actual times spent in phasing orbit and do not include stay times at the satellites, etc. For a time of less than 56.84 hours it is not possible to perform the mission with the phasing orbit approach. It takes 56.84 hours to do one revolution in phasing orbit for each transfer. The first line in each block in Table 3-1 presents the data pertinent to the total effect of all three transfers. The remaining three lines in each

Table 3-1

1 of 7

EFFECT OF TIME ON BASELINE ESO SERVICE MISSION

TIME AVAILABLE (HRS)	TRANSFER	TRANSFER ANGLE (DEG)	NO. PHASE ORBITS	DELTA V (FT/SEC)	PROPELLANT CONSUMED* (LBM)	TIME REQUIRED (HRS)	PHASING ORBIT	
							PERIOD (HRS)	PERIGEE ALT (N.MI.)
56.84-80.78	TOTAL	225.00	3	6541.84	8794.62	56.84	----	----
	SAT. NO. 1 TO 2	35.00	1	725.00	1172.76	21.61	21.61	16321.05
	SAT. NO. 2 TO 3	150.00	1	4975.13	6692.24	13.96	13.96	5577.91
	SAT. NO. 3 TO 4	40.00	1	841.70	929.62	21.27	21.27	15883.82
80.78-104.71	TOTAL	225.00	4	3345.36	4971.09	80.78	----	----
	SAT. NO. 1 TO 2	35.00	1	725.00	1172.76	21.61	21.61	16321.05
	SAT. NO. 2 TO 3	150.00	2	1778.66	2649.37	37.90	18.95	12755.22
	SAT. NO. 3 TO 4	40.00	1	841.70	1148.96	21.27	21.27	15883.82
104.71-128.65	TOTAL	225.00	5	2653.63	4031.57	104.71	----	----
	SAT. NO. 1 TO 2	35.00	1	725.00	1172.76	21.61	21.61	16321.05
	SAT. NO. 2 TO 3	150.00	3	1086.93	1655.95	61.83	20.61	15001.87
	SAT. NO. 3 TO 4	40.00	1	841.70	1202.86	21.27	21.27	15883.82
128.65-152.58	TOTAL	225.00	6	2207.78	3402.77	128.65	----	----
	SAT. NO. 1 TO 2	35.00	1	725.00	1172.76	21.61	21.61	16321.05
	SAT. NO. 2 TO 3	150.00	3	1086.93	1655.95	61.83	20.61	15001.87
	SAT. NO. 3 TO 4	40.00	2	395.85	574.06	45.21	22.60	17619.84
152.58-176.52	TOTAL	225.00	7	1826.59	2850.25	152.58	----	----
	SAT. NO. 1 TO 2	35.00	2	343.81	563.18	45.54	22.77	17834.53
	SAT. NO. 2 TO 3	150.00	3	1086.93	1698.32	61.83	20.61	15001.87
	SAT. NO. 3 TO 4	40.00	2	395.85	588.75	45.21	22.60	17619.84

* NOTE: ISP= 469 SEC. ; W0= 2500 LBM

EFFECT OF TIME ON BASELINE ESO SERVICE MISSION

TIME AVAILABLE (HRS)	TRANSFER	TRANSFER ANGLE (DEG)	NO. PHASE ORBITS	DELTA V (FT/SEC)	PROPELLANT CONSUMED* (LBM)	TIME REQUIRED (HRS)	PHASING ORBIT	
							PERIOD (HRS)	PERIGEE ALT (N.M.I.)
176.52-200.45	TOTAL	225.00	8	1522.60	2399.50	176.52	----	----
	SAT. NO. 1 TO 2	35.00	2	343.81	563.18	45.54	22.77	17834.53
	SAT. NO. 2 TO 3	150.00	4	782.94	1235.59	85.76	21.44	16102.54
	SAT. NO. 3 TO 4	40.00	2	395.85	600.73	45.21	22.60	17619.84
200.45-224.38	TOTAL	225.00	9	1351.48	2141.74	200.45	----	----
	SAT. NO. 1 TO 2	35.00	2	343.81	563.18	45.54	22.77	17834.53
	SAT. NO. 2 TO 3	150.00	5	611.81	970.98	109.70	21.94	16756.46
	SAT. NO. 3 TO 4	40.00	2	395.85	607.58	45.21	22.60	17619.84
224.38-248.32	TOTAL	225.00	10	1214.46	1933.24	224.38	----	----
	SAT. NO. 1 TO 2	35.00	2	343.81	563.18	45.54	22.77	17834.53
	SAT. NO. 2 TO 3	150.00	5	611.81	970.98	109.70	21.94	16756.46
	SAT. NO. 3 TO 4	40.00	3	258.84	399.08	69.14	23.05	18190.93
248.32-272.25	TOTAL	225.00	11	1095.99	1751.43	248.32	----	----
	SAT. NO. 1 TO 2	35.00	3	225.34	370.57	69.48	23.16	18333.46
	SAT. NO. 2 TO 3	150.00	5	611.81	978.63	109.70	21.94	16756.46
	SAT. NO. 3 TO 4	40.00	3	258.84	402.23	69.14	23.05	18190.93
272.25-296.19	TOTAL	225.00	12	986.43	1582.01	272.25	----	----
	SAT. NO. 1 TO 2	35.00	3	225.34	370.57	69.48	23.16	18333.46
	SAT. NO. 2 TO 3	150.00	6	502.25	806.28	133.63	22.27	17188.98
	SAT. NO. 3 TO 4	40.00	3	258.84	405.16	69.14	23.05	18190.93

* NOTE: ISP= 469 SEC. ; W0= 2500 LBM

EFFECT OF TIME ON BASELINE ESO SERVICE MISSION

TIME AVAILABLE (HRS)	TRANSFER	TRANSFER ANGLE (DEG)	NO. PHASE ORBITS	DELTA V (FT/SEC)	PROPELLANT CONSUMED* (LBM)	TIME REQUIRED (HRS)	PHASING ORBIT	
							PERIOD (HRS)	PERIGEE ALT (N.M.I.)
296.19-320.12	TOTAL	225.00	13	910.05	1463.18	296.19	----	----
	SAT. NO. 1 TO 2	35.00	3	225.34	370.57	69.48	23.16	18333.46
	SAT. NO. 2 TO 3	150.00	7	425.87	685.40	157.57	22.51	17497.16
	SAT. NO. 3 TO 4	40.00	3	258.84	407.22	69.14	23.05	18190.93
320.12-344.06	TOTAL	225.00	14	843.55	1359.21	320.12	----	----
	SAT. NO. 1 TO 2	35.00	3	225.34	370.57	69.48	23.16	18333.46
	SAT. NO. 2 TO 3	150.00	7	425.87	685.40	157.57	22.51	17497.16
	SAT. NO. 3 TO 4	40.00	4	192.33	303.25	93.08	23.27	18474.98
344.06-367.99	TOTAL	225.00	15	785.88	1268.70	344.06	----	----
	SAT. NO. 1 TO 2	35.00	4	167.68	276.27	93.41	23.35	18581.45
	SAT. NO. 2 TO 3	150.00	7	425.87	688.02	157.57	22.51	17497.16
	SAT. NO. 3 TO 4	40.00	4	192.33	304.41	93.08	23.27	18474.98
367.99-391.92	TOTAL	225.00	16	729.80	1180.33	367.99	----	----
	SAT. NO. 1 TO 2	35.00	4	167.68	276.27	93.41	23.35	18581.45
	SAT. NO. 2 TO 3	150.00	8	369.79	598.52	181.50	22.69	17727.05
	SAT. NO. 3 TO 4	40.00	4	192.33	305.54	93.08	23.27	18474.98
391.92-415.86	TOTAL	225.00	17	686.68	1112.17	391.92	----	----
	SAT. NO. 1 TO 2	35.00	4	167.68	276.27	93.41	23.35	18581.45
	SAT. NO. 2 TO 3	150.00	9	326.67	529.48	205.44	22.83	17905.86
	SAT. NO. 3 TO 4	40.00	4	192.33	306.42	93.08	23.27	18474.98

* NOTE: ISP= 469 SEC. ; W0= 250 LBM

EFFECT OF TIME ON BASELINE ESO SERVICE MISSION

TIME AVAILABLE (HRS)	TRANSFER	TRANSFER ANGLE (DEG)	NO. PHASE ORBITS	DELTA V (FT/SEC)	PROPELLANT CONSUMED* (LBM)	TIME REQUIRED (HRS)	PHASING ORBIT	
							PERIOD (HRS)	PERIGEE (N.M.I.)
415.86-439.79	TOTAL	225.00	18	647.27	1049.70	415.86	-----	-----
	SAT. NO. 1 TO 2	35.00	4	167.68	276.27	93.41	23.35	18581.45
	SAT. NO. 2 TO 3	150.00	9	326.67	529.48	205.44	22.83	17905.86
	SAT. NO. 3 TO 4	40.00	5	152.92	243.95	117.01	23.40	18645.50
439.79-463.73	TOTAL	225.00	19	613.12	995.44	439.79	-----	-----
	SAT. NO. 1 TO 2	35.00	5	133.53	220.25	117.34	23.47	18730.01
	SAT. NO. 2 TO 3	150.00	9	326.67	530.68	205.44	22.83	17995.86
	SAT. NO. 3 TO 4	40.00	5	152.92	244.51	117.01	23.40	18645.50
463.73-487.66	TOTAL	225.00	20	579.06	941.20	463.73	-----	-----
	SAT. NO. 1 TO 2	35.00	5	133.53	220.25	117.34	23.47	18730.01
	SAT. NO. 2 TO 3	150.00	10	292.61	475.89	229.37	22.94	18048.40
	SAT. NO. 3 TO 4	40.00	5	152.92	245.06	117.01	23.40	18645.50
487.66-511.60	TOTAL	225.00	21	551.39	897.04	487.66	-----	-----
	SAT. NO. 1 TO 2	35.00	5	133.53	220.25	117.34	23.47	18730.01
	SAT. NO. 2 TO 3	150.00	11	264.94	431.27	253.31	23.03	18165.11
	SAT. NO. 3 TO 4	40.00	5	152.92	245.51	117.01	23.40	18645.50
511.60-535.53	TOTAL	225.00	22	525.48	855.61	511.60	-----	-----
	SAT. NO. 1 TO 2	35.00	5	133.53	220.25	117.34	23.47	18730.01
	SAT. NO. 2 TO 3	150.00	11	264.94	431.27	253.31	23.03	18165.11
	SAT. NO. 3 TO 4	40.00	6	127.01	204.08	140.95	23.49	18753.54

* NOTE: ISP= 469 SEC. ; W0= 250. LBM

EFFECT OF TIME ON BASELINE ESO SERVICE MISSION

TIME AVAILABLE (HRS)	TRANSFER	TRANSFER ANGLE (DEG)	NO. PHASE ORBITS	DELTA V PROPELLANT (FT/SEC) CONSUMED* (LBM)	TIME REQUIRED (HRS)	PHASING ORBIT	
						PERIOD (HRS)	PERIGEE ALT (N.MI.)
535.53-559.46	TOTAL	225.00	23	502.66	535.53	-----	-----
	SAT. NO. 1 TO 2	35.00	5	133.53	117.34	23.47	18730.31
	SAT. NO. 2 TO 3	150.00	12	242.12	277.24	23.10	18261.88
	SAT. NO. 3 TO 4	40.00	6	127.01	140.95	23.49	18758.54
559.46-583.40	TOTAL	225.00	24	480.06	559.46	-----	-----
	SAT. NO. 1 TO 2	35.00	6	110.94	141.28	23.55	18828.98
	SAT. NO. 2 TO 3	150.00	12	242.12	277.24	23.10	18261.88
	SAT. NO. 3 TO 4	40.00	6	127.01	140.95	23.49	18758.54
583.40-607.33	TOTAL	225.00	25	460.85	583.40	-----	-----
	SAT. NO. 1 TO 2	35.00	6	110.94	141.28	23.55	18828.98
	SAT. NO. 2 TO 3	150.00	13	222.91	301.17	23.17	18343.83
	SAT. NO. 3 TO 4	40.00	6	127.01	140.95	23.49	18758.54
607.33-631.27	TOTAL	225.00	26	442.50	607.33	-----	-----
	SAT. NO. 1 TO 2	35.00	6	110.94	141.28	23.55	18828.98
	SAT. NO. 2 TO 3	150.00	13	222.91	301.17	23.17	18343.83
	SAT. NO. 3 TO 4	40.00	7	108.66	164.88	23.56	18838.96
631.27-655.20	TOTAL	225.00	27	425.99	631.27	-----	-----
	SAT. NO. 1 TO 2	35.00	6	110.94	141.28	23.55	18828.98
	SAT. NO. 2 TO 3	150.00	14	206.40	325.11	23.21	18414.57
	SAT. NO. 3 TO 4	40.00	7	108.66	164.88	23.56	18838.96

* NOTE: ISP= 469 SEC. ; W0= 250 LBM

EFFECT OF TIME ON BASELINE ESO SERVICE MISSION

TIME AVAILABLE (HRS)	TRANSFER	TRANSFER ANGLE (DEG)	NO. PHASE ORBITS	DELTA V (FT/SEC)	PROPELLANT CONSUMED* (LBM)	TIME REQUIRED (HRS)	PHASING ORBIT	
							PERIOD (HRS)	PERIGEE ALT (N.M.I.)
655.20-679.14	TOTAL	225.00	28	409.96	670.07	655.20	----	----
	SAT. NO. 1 TO 2	35.00	7	94.91	156.75	165.21	23.60	18899.55
	SAT. NO. 2 TO 3	150.00	14	206.40	337.50	325.11	23.27	18474.98
	SAT. NO. 3 TO 4	40.00	7	108.66	175.83	164.88	23.55	18838.96
679.14-703.07	TOTAL	225.00	29	395.89	647.37	679.14	----	----
	SAT. NO. 1 TO 2	35.00	7	94.91	156.75	165.21	23.60	18899.55
	SAT. NO. 2 TO 3	150.00	15	192.33	314.63	349.04	23.27	18474.98
	SAT. NO. 3 TO 4	40.00	7	108.66	175.99	164.88	23.55	18838.96
703.07-727.01	TOTAL	225.00	30	382.14	625.17	703.07	----	----
	SAT. NO. 1 TO 2	35.00	7	94.91	156.75	165.21	23.60	18899.55
	SAT. NO. 2 TO 3	150.00	15	192.33	314.63	349.04	23.27	18474.98
	SAT. NO. 3 TO 4	40.00	8	94.91	153.79	188.82	23.60	18899.55
727.01-750.94	TOTAL	225.00	31	369.75	605.15	727.01	----	----
	SAT. NO. 1 TO 2	35.00	7	94.91	156.75	165.21	23.60	18899.55
	SAT. NO. 2 TO 3	150.00	16	179.94	294.49	372.98	23.31	18528.45
	SAT. NO. 3 TO 4	40.00	8	94.91	153.92	188.82	23.60	18899.55
750.94-774.87	TOTAL	225.00	32	357.75	585.74	750.94	----	----
	SAT. NO. 1 TO 2	35.00	8	82.91	136.98	189.15	23.64	18952.51
	SAT. NO. 2 TO 3	150.00	16	179.94	294.72	372.98	23.31	18528.45
	SAT. NO. 3 TO 4	40.00	8	94.91	154.04	188.82	23.60	18899.55

* NOTE: ISP= 469 SEC. ; W0= 25000 LBM

EFFECT OF TIME ON BASELINE ESO SERVICE MISSION

TIME AVAILABLE (HRS)	TRANSFER	TRANSFER ANGLE (DEG)	NO. PHASE ORBITS	DELTA V (FT/SEC)	PROPELLANT CONSUMED* (LBM)	TIME REQUIRED (HRS)	PHASING ORBIT	
							PERIOD (HRS)	PERIGEE ALT (N.M.I.)
774.87-798.81	TOTAL	225.00	33	346.91	568.20	774.87	---	---
	SAT. NO. 1 TO 2	35.00	8	82.91	136.98	189.15	23.64	18952.51
	SAT. NO. 2 TO 3	150.00	17	169.10	277.07	396.91	23.35	18575.30
	SAT. NO. 3 TO 4	40.00	8	94.91	154.15	188.82	23.60	18899.55
798.81-822.74	TOTAL	225.00	34	336.20	590.85	798.81	---	---
	SAT. NO. 1 TO 2	35.00	8	82.91	136.98	189.15	23.64	18952.51
	SAT. NO. 2 TO 3	150.00	17	169.10	277.07	396.91	23.35	18575.30
	SAT. NO. 3 TO 4	40.00	9	84.20	136.80	212.75	23.64	18946.78
822.74-846.68	TOTAL	225.00	35	326.59	535.27	822.74	---	---
	SAT. NO. 1 TO 2	35.00	8	82.91	136.98	189.15	23.64	18952.51
	SAT. NO. 2 TO 3	150.00	18	159.48	261.40	420.85	23.38	18617.01
	SAT. NO. 3 TO 4	40.00	9	84.20	136.89	212.75	23.64	18946.78
846.68-870.61	TOTAL	225.00	36	317.30	520.21	846.68	---	---
	SAT. NO. 1 TO 2	35.00	9	73.62	121.68	213.08	23.68	18993.63
	SAT. NO. 2 TO 3	150.00	18	159.48	261.56	420.85	23.38	18617.01
	SAT. NO. 3 TO 4	40.00	9	84.20	136.97	212.75	23.64	18946.78
870.61-894.55	TOTAL	225.00	37	308.78	506.38	870.61	---	---
	SAT. NO. 1 TO 2	35.00	9	73.62	121.68	213.08	23.68	18993.63
	SAT. NO. 2 TO 3	150.00	19	150.96	247.65	444.78	23.41	18654.00
	SAT. NO. 3 TO 4	40.00	9	84.20	137.05	212.75	23.64	18946.78

* NOTE: ISP= 469 SEC. ; W0= 250 LBM

block give a breakdown for each of the three transfers respectively. The data presented include transfer angle, number of revolutions in phasing orbit, ΔV , propellant consumed, and time required. In addition the period and perigee altitude are given for each of the three phasing orbits.

In the first block where the total number of phasing orbit revolutions is 3, it is uniquely determined that one will have to be assigned to each of the three transfers. In the second block where the total number of phasing orbit revolutions is 4, the possible splits include (2,1,1), (1,2,1), and (1,1,2). The (1,2,1) split has been found to result in the minimum total ΔV and the data presented in the second block reflects that split.

The table could be used in the following way. Suppose there are 350 hours available for the total of all three transfers. Then one would find the block with a time available of 344.06-367.99 hours. The total number of revolutions in phasing orbit would be 15 with a split of (4,7,4). The actual time used would only be 344.06 hours. The extra 6 hours would not be enough to allow for an additional revolution in phasing orbit and thus would not be used.

Loiter Missions

As part of the study effort, preliminary analysis of the effects of "loitering" in circular parking orbits from 160 to 1000 n.mi. in altitude was performed. This analysis addressed itself primarily to determining the effect of orbit altitude on Auxiliary Propulsion System (APS) propellant consumption. The effect of boiloff was not

initially considered but was brought into account as the work progressed. As a first cut this analysis made a number of simplifying assumptions. These assumptions are noted on the data and the figures described.

Figure 3-4 presents the effect of orbit altitude on vehicle drag. In the absence of aerodynamic data on the Tug, drag coefficients generated for the undeployed Skylab were used with appropriate adjustment of reference area. An angle of attack of 5° in both the pitch and yaw planes was assumed. Notice there are three curves labeled "BEST SUN," "MEAN SUN," and "WORST SUN." Drag is proportional to the atmospheric density which is in turn affected and influenced by sunspot activity. Sunspot activity projections and estimates vary over an 11.1 year cycle with a wide dispersion around the mean. "BEST SUN" reflects -2σ estimates of solar activity at the least active point in the 11.1 year cycle. "MEAN SUN" reflects the projected average solar activity over the entire cycle. "WORST SUN" reflects $+2\sigma$ estimates of solar activity at the most active point in the solar cycle.

The estimated propellant consumption rate in order to keep the loiter orbit from decaying is presented in Figure 3-5. As with drag, this data reflects the effects of solar activity. Notice that the difference between "MEAN SUN" and "WORST SUN" is much greater than the difference between "MEAN SUN" and "BEST SUN." The fact that the average solar activity is much closer to the minimum than to the maximum will be seen repeatedly throughout the

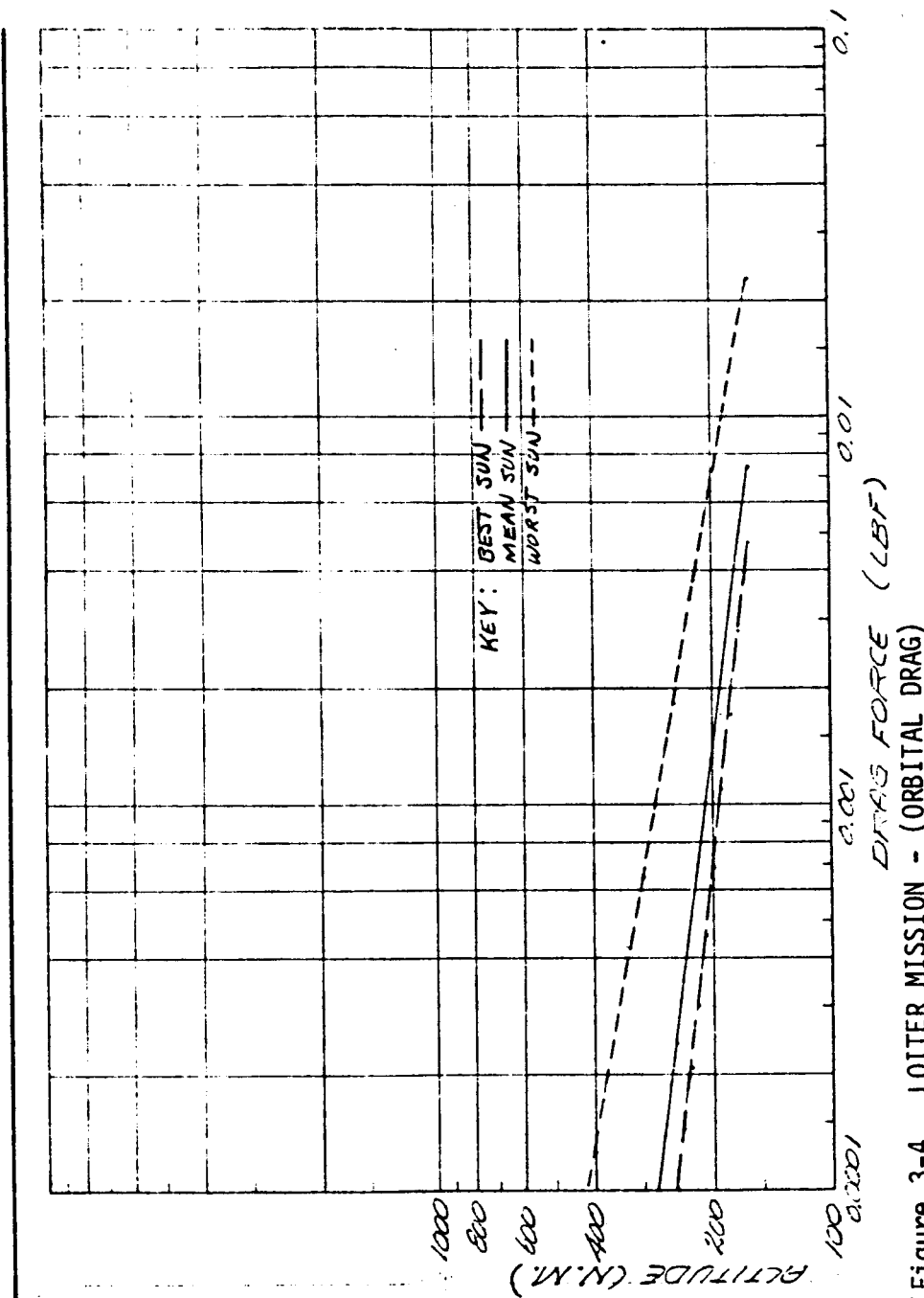


Figure 3-4. LOITER MISSION - (ORBITAL DRAG)

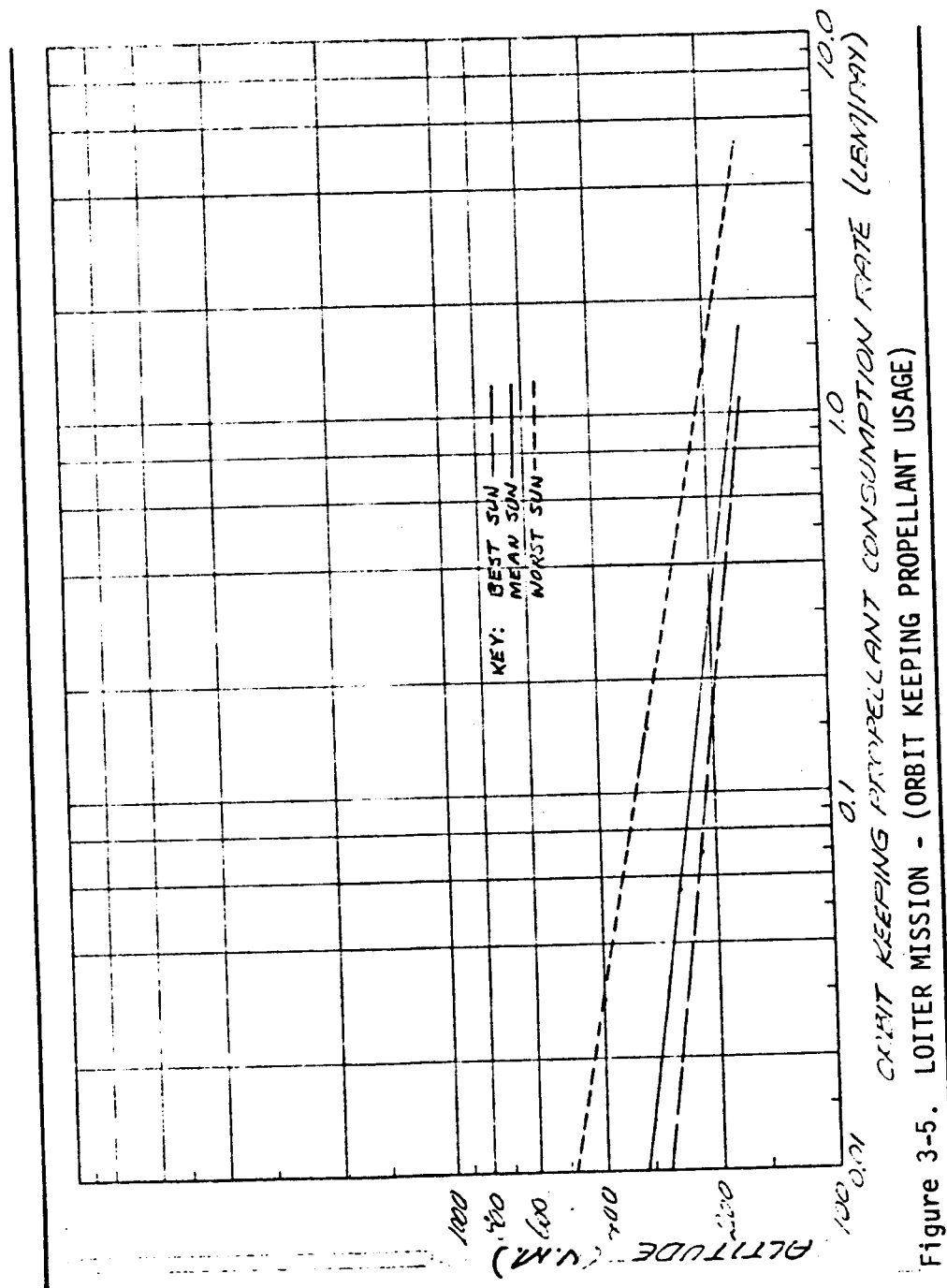


Figure 3-5. LOITER MISSION - (ORBIT KEEPING PROPELLANT USAGE)

data presented in this analysis. At the 160 n.mi. worst case altitude orbit keeping only cost 5 lbm/day. Propellant consumption quickly drops to less than 1 lbm/day for "WORST SUN" conditions above 210 n.mi.

The drag and orbit keeping propellant previously discussed were unaffected by vehicle mass characteristics. Most other parameters of interest are affected. Therefore, some assumptions have to be made concerning propellant remaining in the tanks and payload weight carried. In order to define the range of possible loiter missions three specific cases were analyzed.

- (1) Pre-Mission Loiter - The Tug loiters without a payload and with full tanks (except for the propellant used to get from 160 n.mi. to the loiter altitude).
- (2) Payload Attendance Loiter - Same as pre-mission loiter but assumes 5000 lbm payload.
- (3) Post-Mission Loiter - The Tug loiters with 3000 lbm payload with only enough propellant left to go from loiter orbit to 160 n.mi.

Figures 3-6 through 3-8 now present the expected orbital altitude decay times to 100 n.mi. for the three mission types analyzed assuming orbit keeping was not attempted. The pre-mission loiter and payload attendance loiter decay times are very close. They both take about 80 days to reach 100 n.mi. starting at 160 n.mi. for "WORST SUN." However, for the post-mission loiter, the decay problem can be much more severe. The expected "WORST SUN" decay

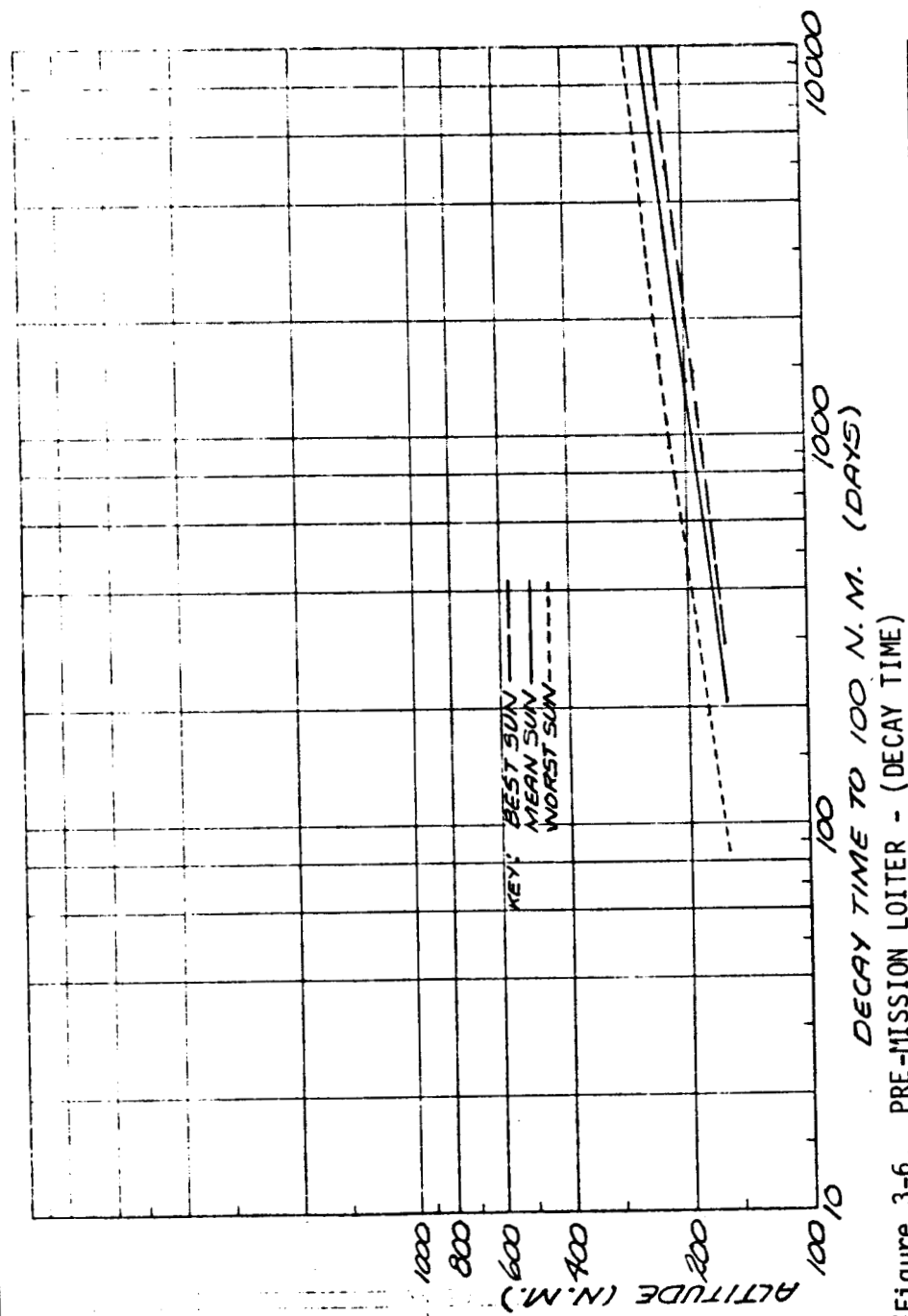


Figure 3-6. PRE-MISSION LOITER - (DECAY TIME)

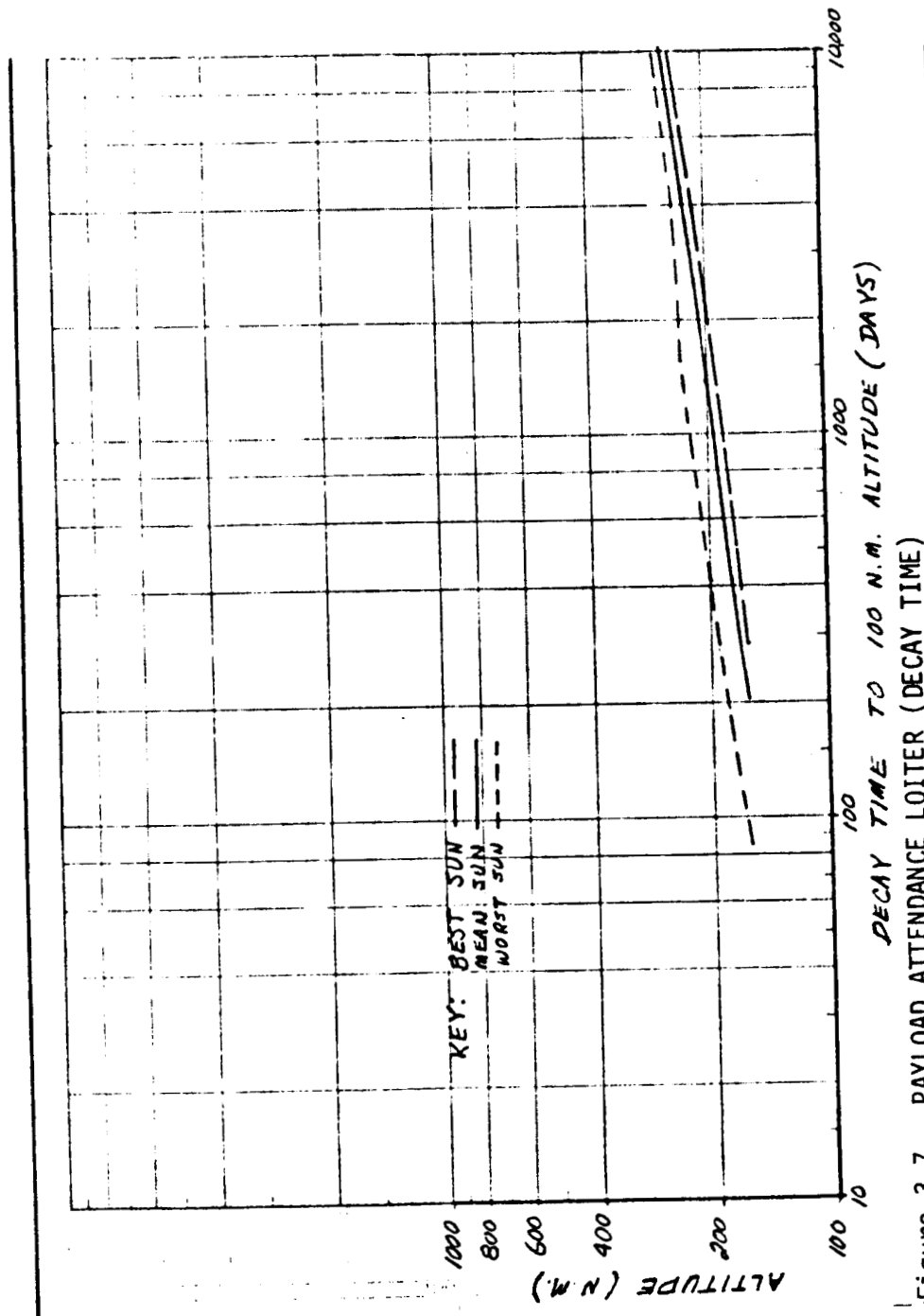


Figure 3-7. PAYLOAD ATTENDANCE LOITER (DECAY TIME)

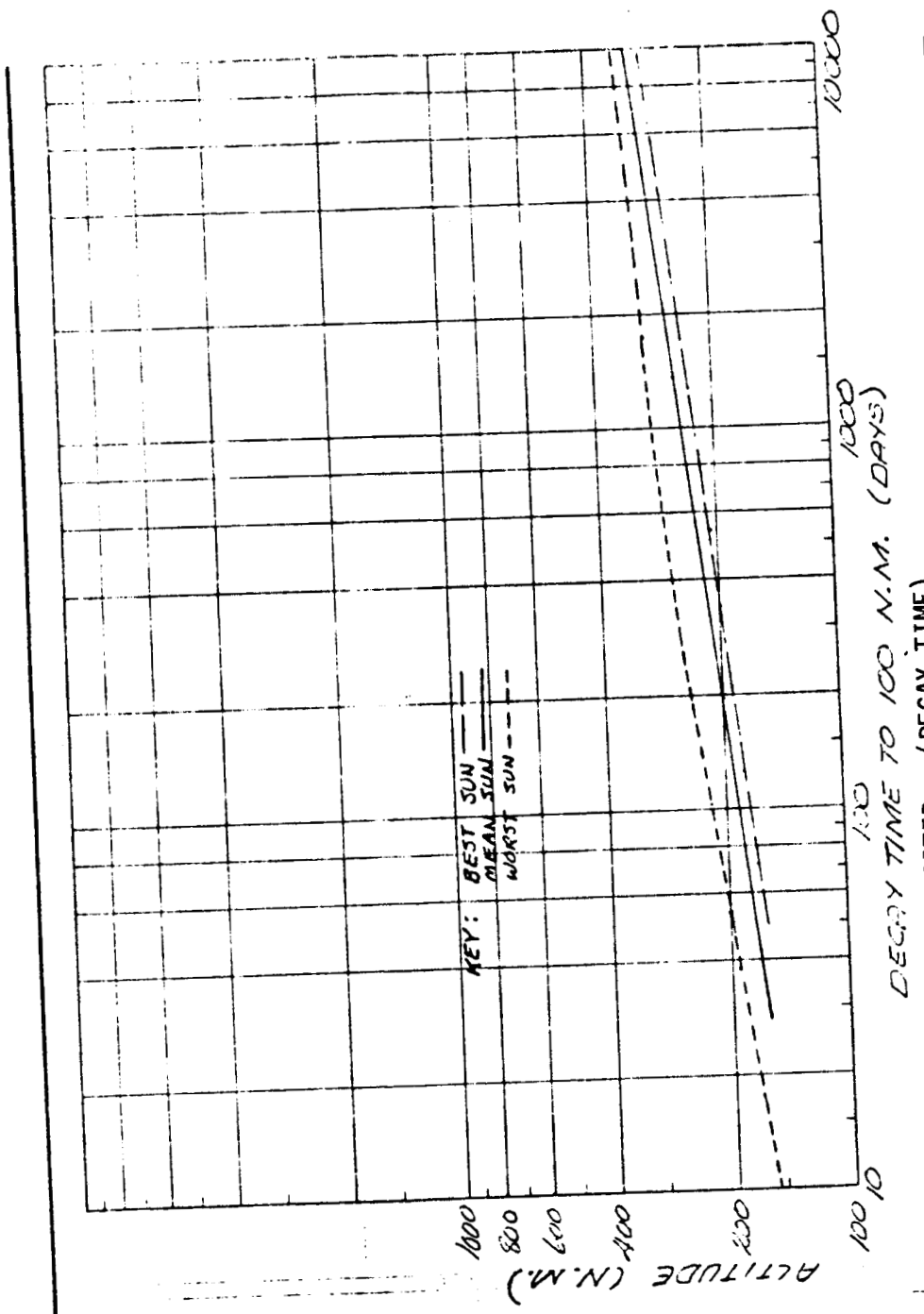


Figure 3-8. POST-MISSION LOITER - (DECAY TIME)

time is reduced to 10 days. It should be noted that in any case decay problems can be minimized by a slight increase in loiter altitude. This would be a particularly feasible approach for post-mission loiter since the Tug will probably be coming back from higher orbit mission anyway.

Figures 3-9 through 3-11 present the orbit altitude effect on external torque forces for the three mission types. The gravity gradient torque plotted is in the pitch plane. The effect in yaw and roll is assumed to be insignificantly small. The aerodynamic torque is not a function of mass characteristics and therefore appears the same on all three figures. The gravity gradient torque, however, is proportional to the difference between pitch and roll moments of inertia. The 5000 lbm payload for the payload attendance loiter increases the pitch moment of inertia and therefore results in an increased gravity gradient torque for this mission over the others. The aerodynamic torque plotted in Figures 3-9 through 3-11 is in fact the torque in both the pitch and yaw planes, since the angle of attack in both planes is assumed to be 5° . Note that the aerodynamic torque falls off very rapidly with increasing altitude by almost an order of magnitude with just a 100 n.mi. altitude increase. Gravity gradient torque is much less sensitive to increasing altitude. At 1000 n.mi. for the three missions it is never less than 50 percent of its value at 160 n.mi.

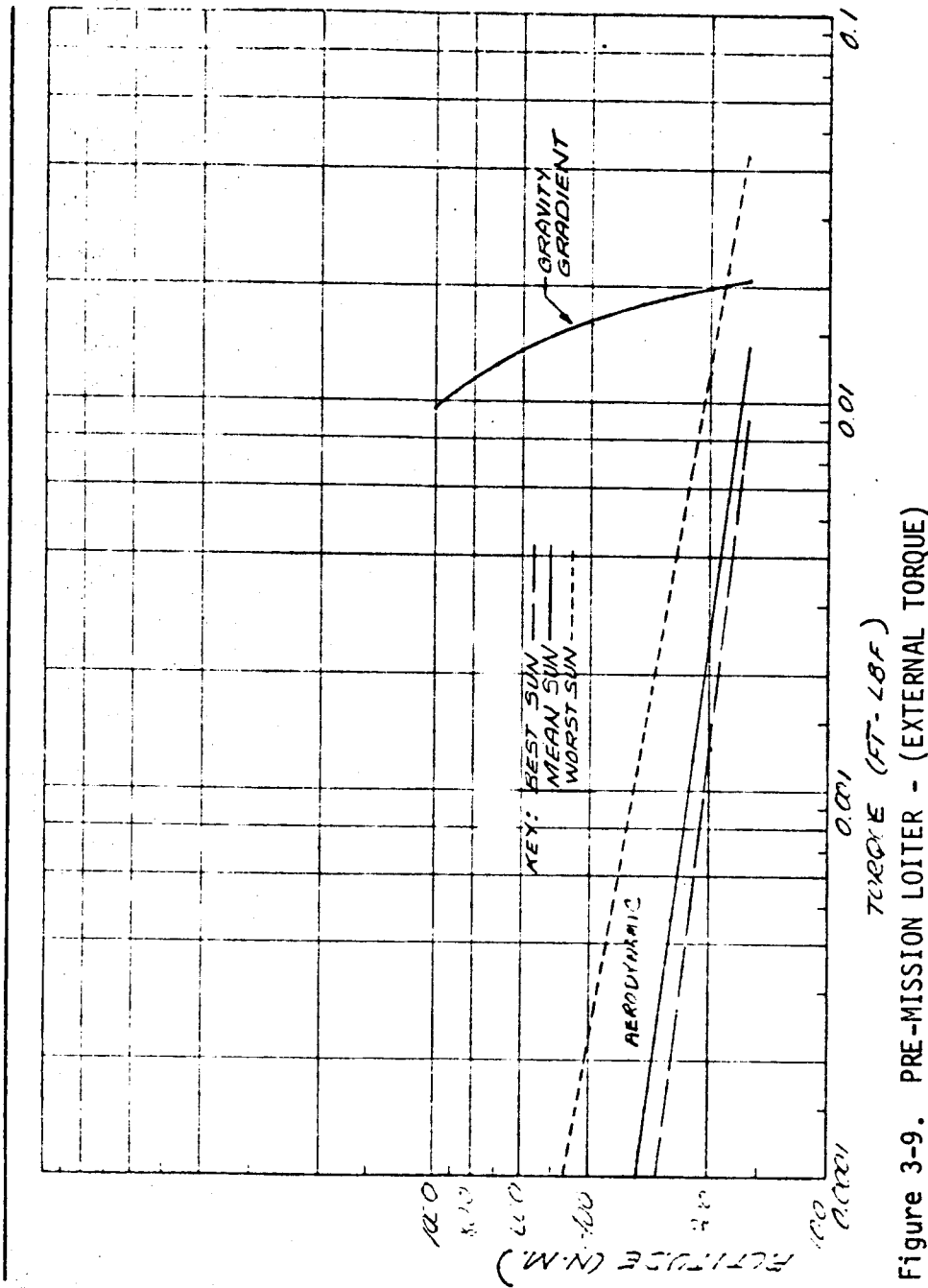


Figure 3-9. PRE-MISSION LOITER - (EXTERNAL TORQUE)

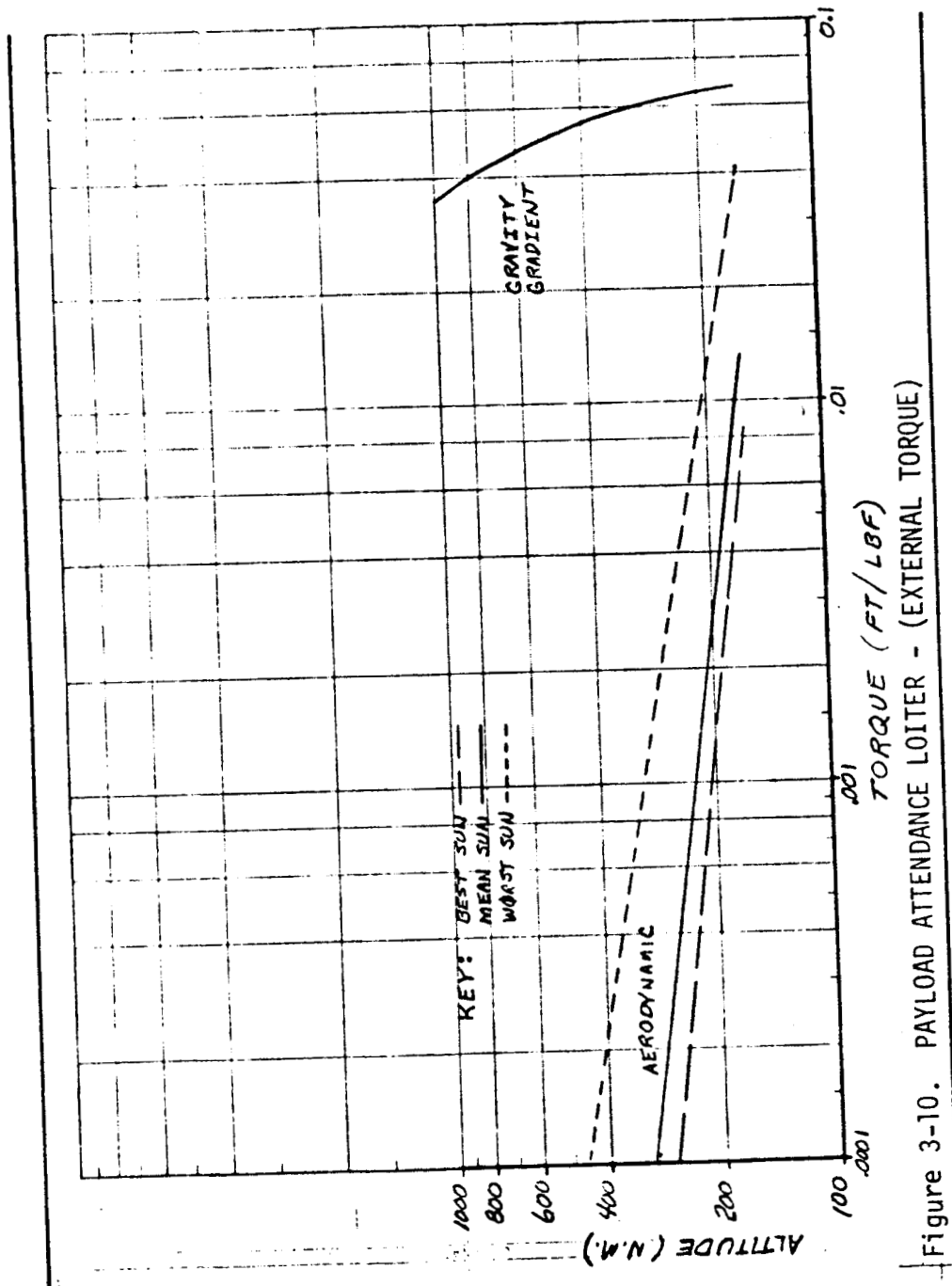


Figure 3-10. PAYLOAD ATTENDANCE LOITER - (EXTERNAL TORQUE)

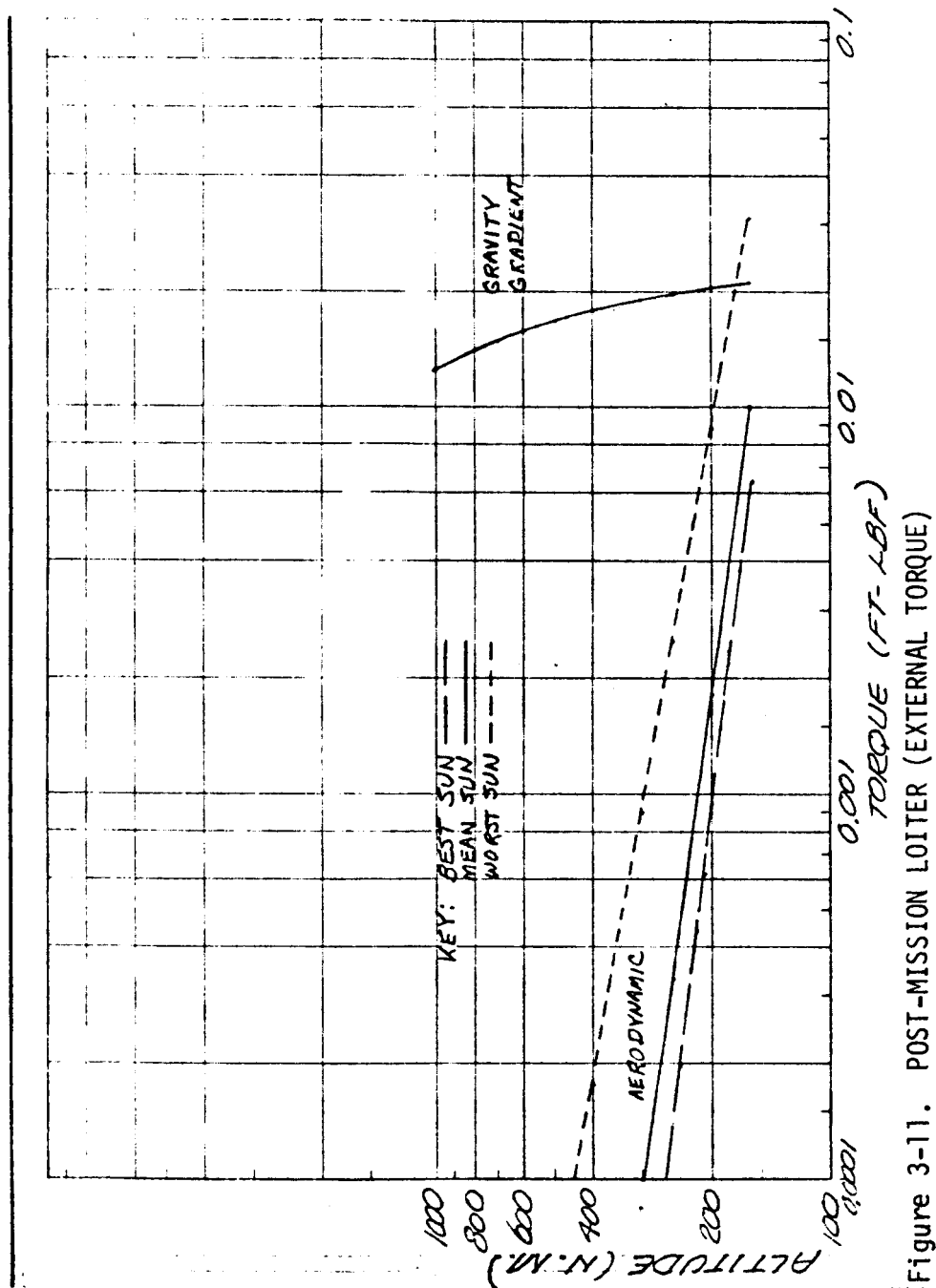


Figure 3-11. POST-MISSION LOITER (EXTERNAL TORQUE)

Figures 3-12 through 3-14 present propellant consumption rates for the pitch, yaw, and roll axes control as function of altitude for the three mission types. Roll control was assumed to be achieved by a two-sided unperturbed limit cycle with a deadband of 5° and a minimum thrust pulse width of 30 ms. Roll thrusters were assumed to be 25 lbf with a separation distance of 15 feet. Control in the pitch plane and in the yaw plane was achieved with a perturbed one-sided limit cycle with a deadband of 5° . Pitch and yaw thrusters were assumed to be 100 lbf with a separation distance of 15 feet. Under these assumptions the required propellant for pitch and yaw control are directly portioned to the external torque. The backward "S" shape of the plots for pitch control flowrate can be accounted for by summing the aerodynamic and gravity gradient torques in the pitch plane. The yaw control flowrate is not affected by gravity gradient torque and thus assumes the shape of the aerodynamic torque only.

The differences in roll control propellant rates for the three loiter mission types reflect differences in the values for roll moment of inertia. The increased pitch control flowrate for the payload attendance mission reflects the increased gravity gradient torque previously noted. The effect of the sun on the pitch flowrate disappears above about 400 n.mi. because the aerodynamic torque becomes vanishingly small compared to the gravity gradient torque. The ever decreasing yaw control flowrate as a function of increasing altitude is somewhat misleading because at some point the external torque will become low enough that control will have to switch from

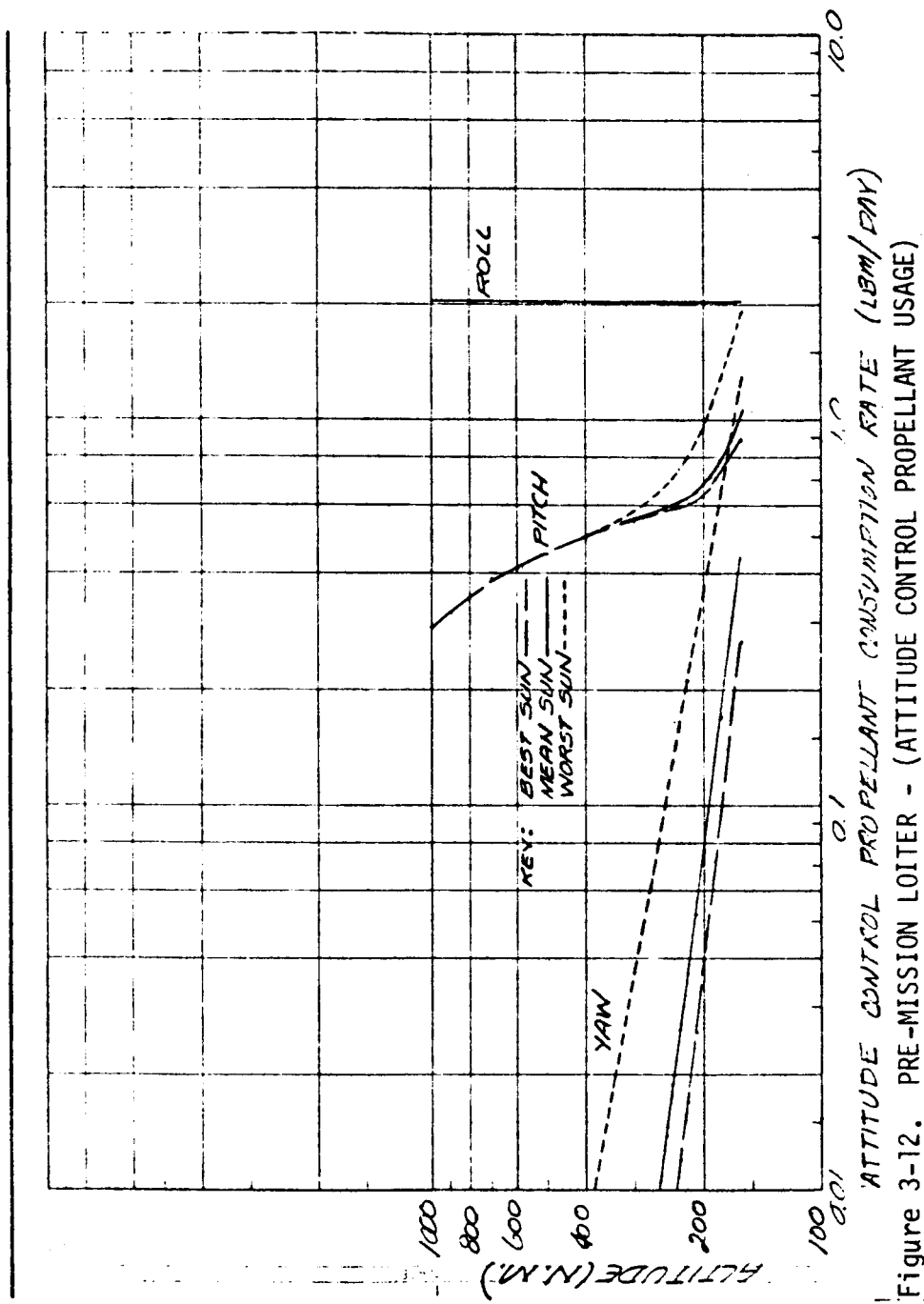


Figure 3-12. PRE-MISSION LOITER - (ATTITUDE CONTROL PROPELLANT USAGE)

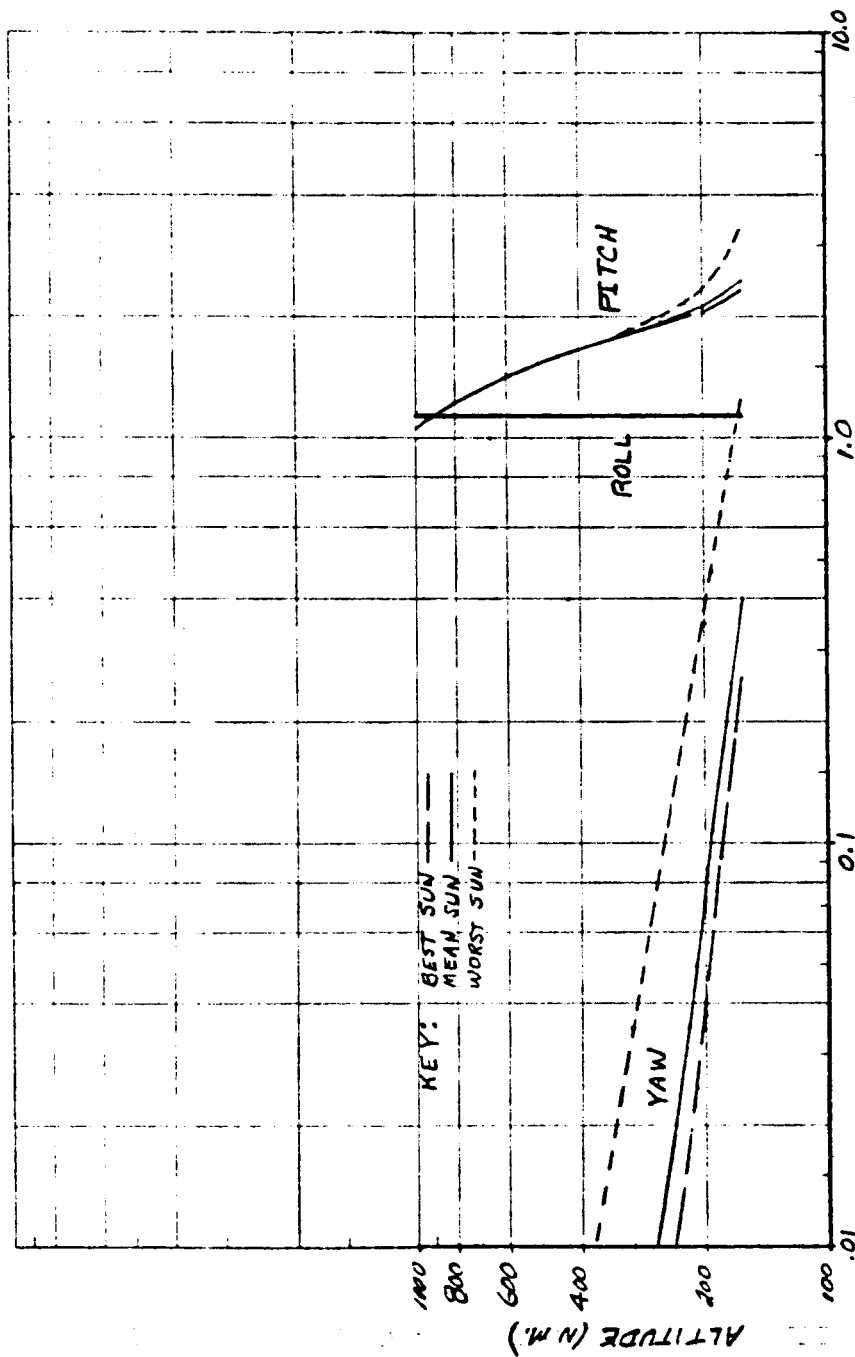


Figure 3-13. PAYLOAD ATTENDANCE LOITER - (ATTITUDE CONTROL PROPELLANT USAGE)

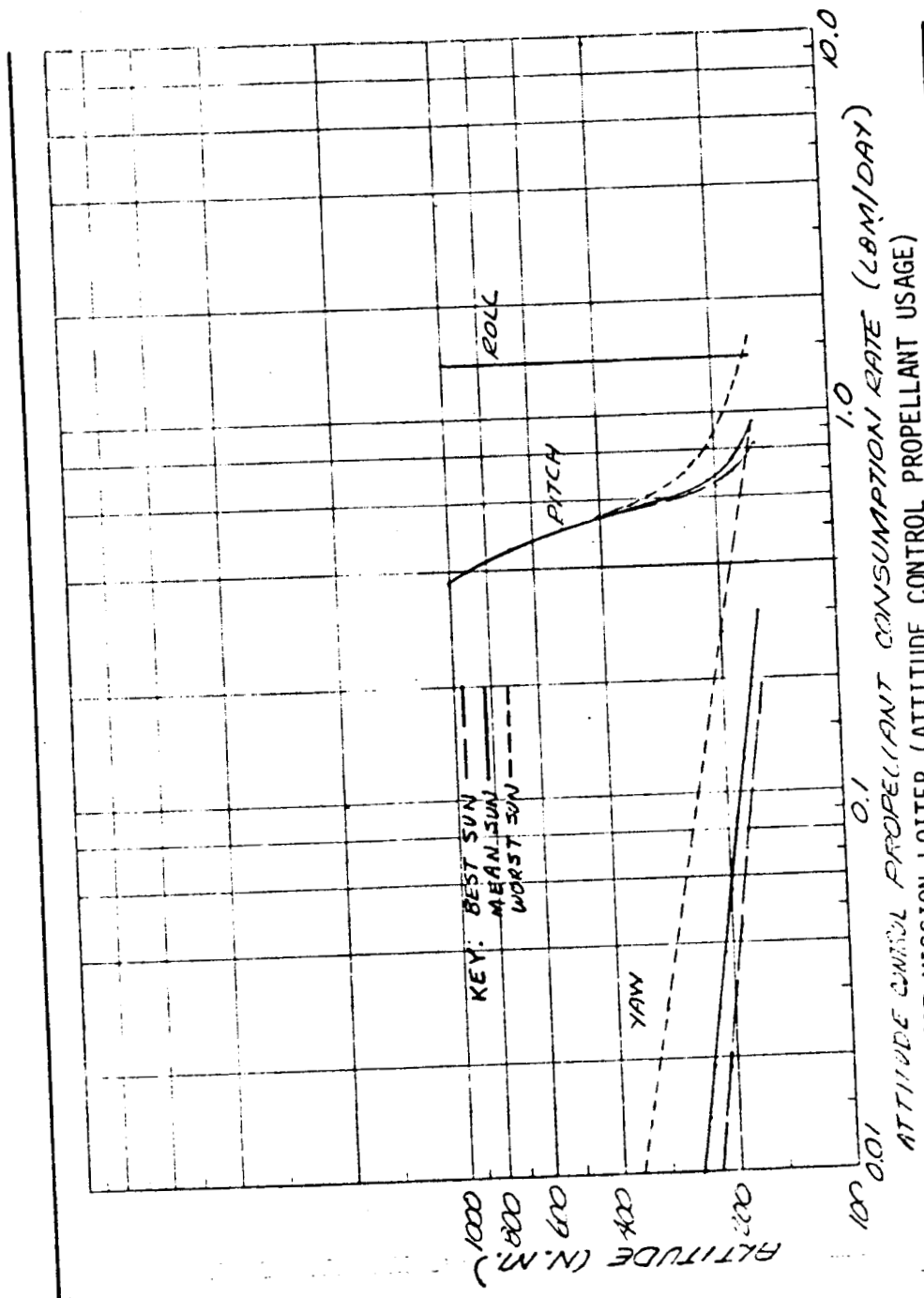


Figure 3-14. POST-MISSION LOITER (ATTITUDE CONTROL PROPELLANT USAGE)

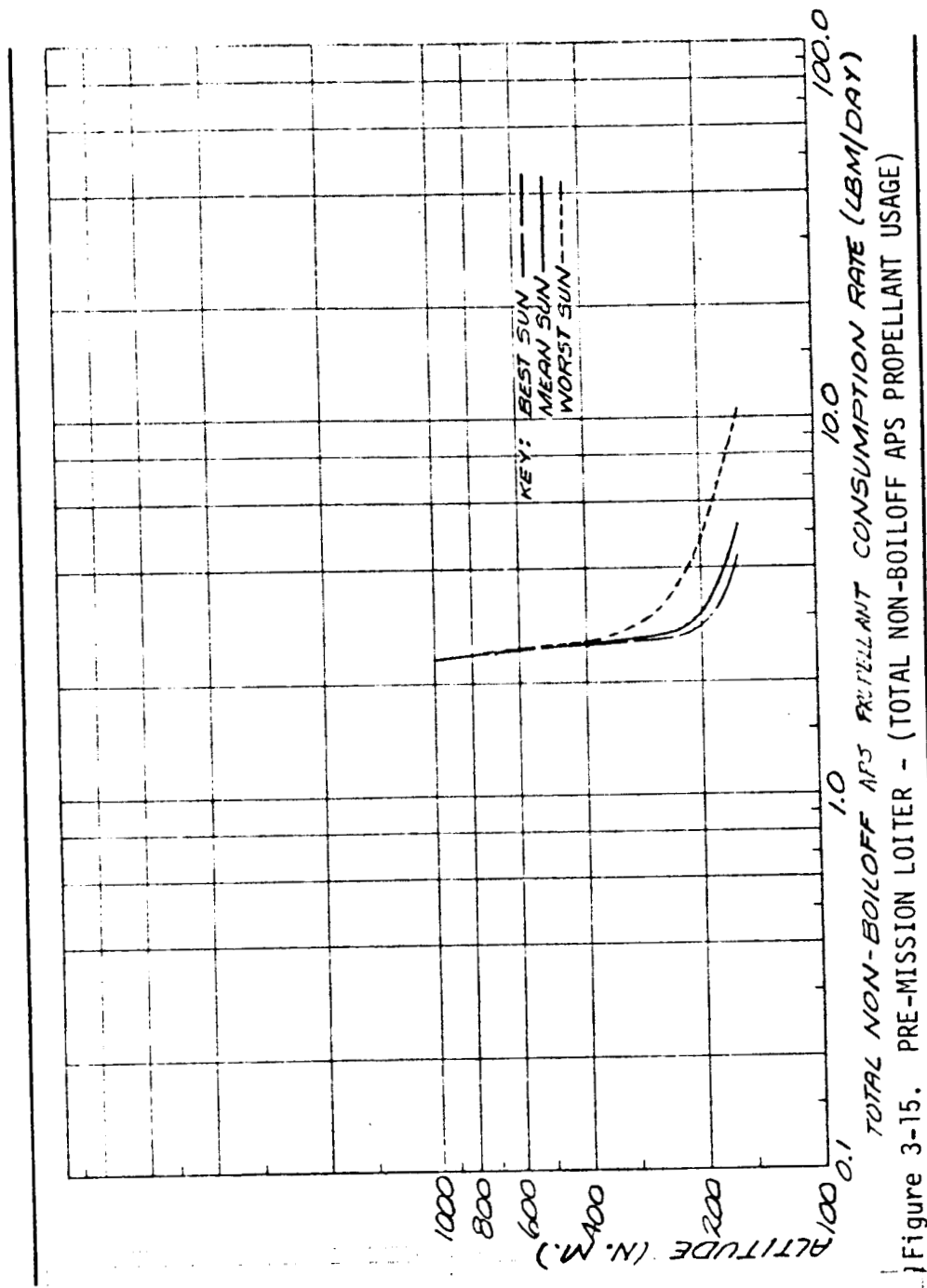
the one-sided limit cycle to a two-sided perturbed limit cycle at which point the propellant usage will go up by an order of magnitude or more. Consideration of this effect will have to await a more in-depth study.

Figures 3-15 through 3-17 present the sum of all the attitude control and orbit keeping propellant consumption rates as a function of altitude for the three mission types respectively. The most notable thing about these three figures is that the mission type has very little effect on total propellant consumption rate. Non-boiloff propellant consumption varies from 2 to 10 lbm/day depending on altitude, solar activity, etc. For loiter orbits above 400 n.mi. atmospheric effects are minimal and non-boiloff APS propellant consumption rates range from 2 to 3 lbm/day.

Additional Missions

In addition to the geosynchronous and the loiter missions there are at least two other mission types which offer definite academic interest and scientific promise for an extended Tug, i.e., lunar and libration point missions. In addition, technical applications of these missions could emerge as time progresses.

It is estimated that most lunar missions will be from 14 to 28 days in duration. The 28,000 fps velocity requirement provides sufficient budget for earth orbit to lunar orbit and back. The Tug delivery capability of 4,000 to 10,000 lbm, depending on how much weight is to be returned, opens up a wide range of unmanned lunar mission possibilities.



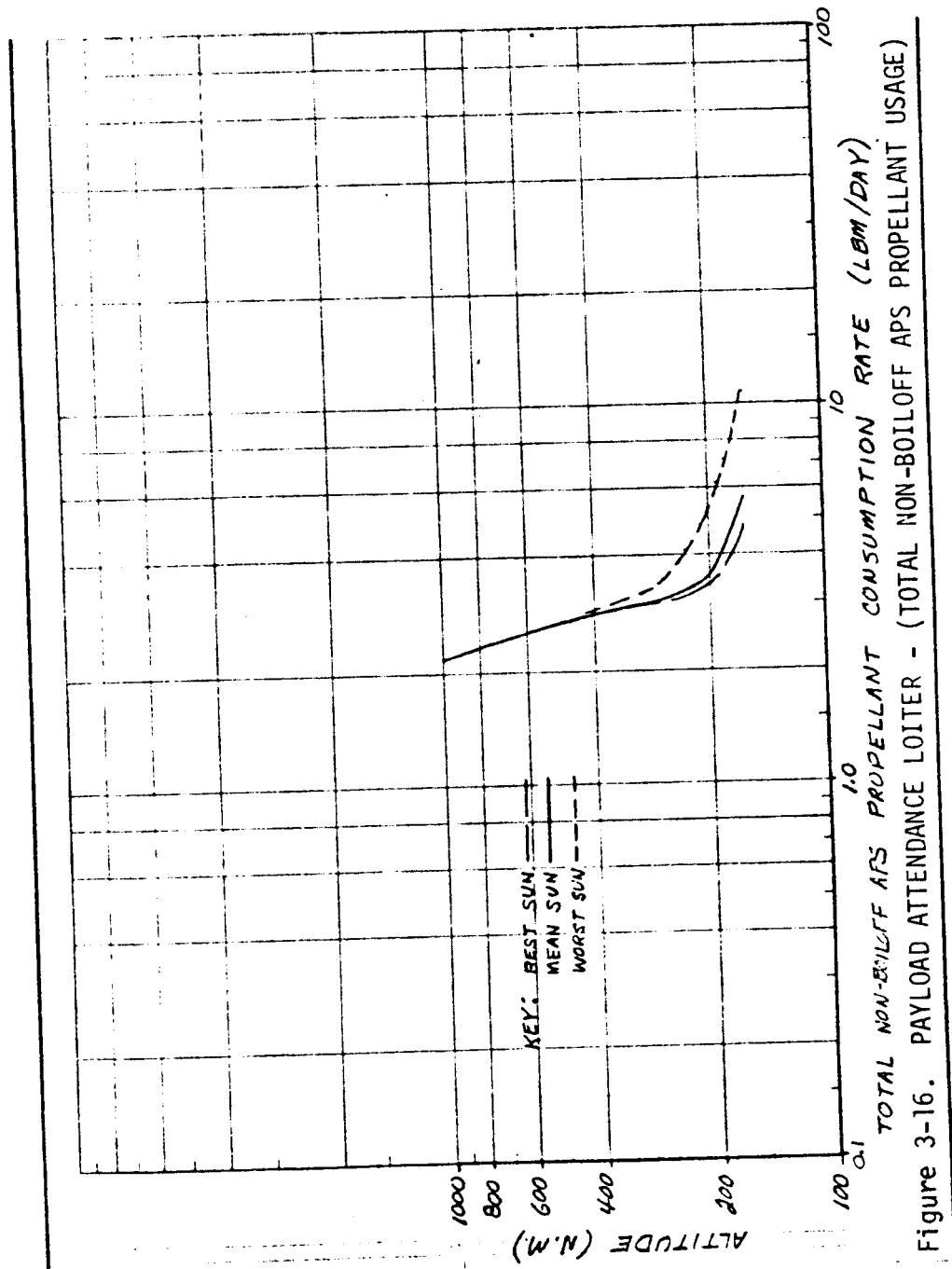
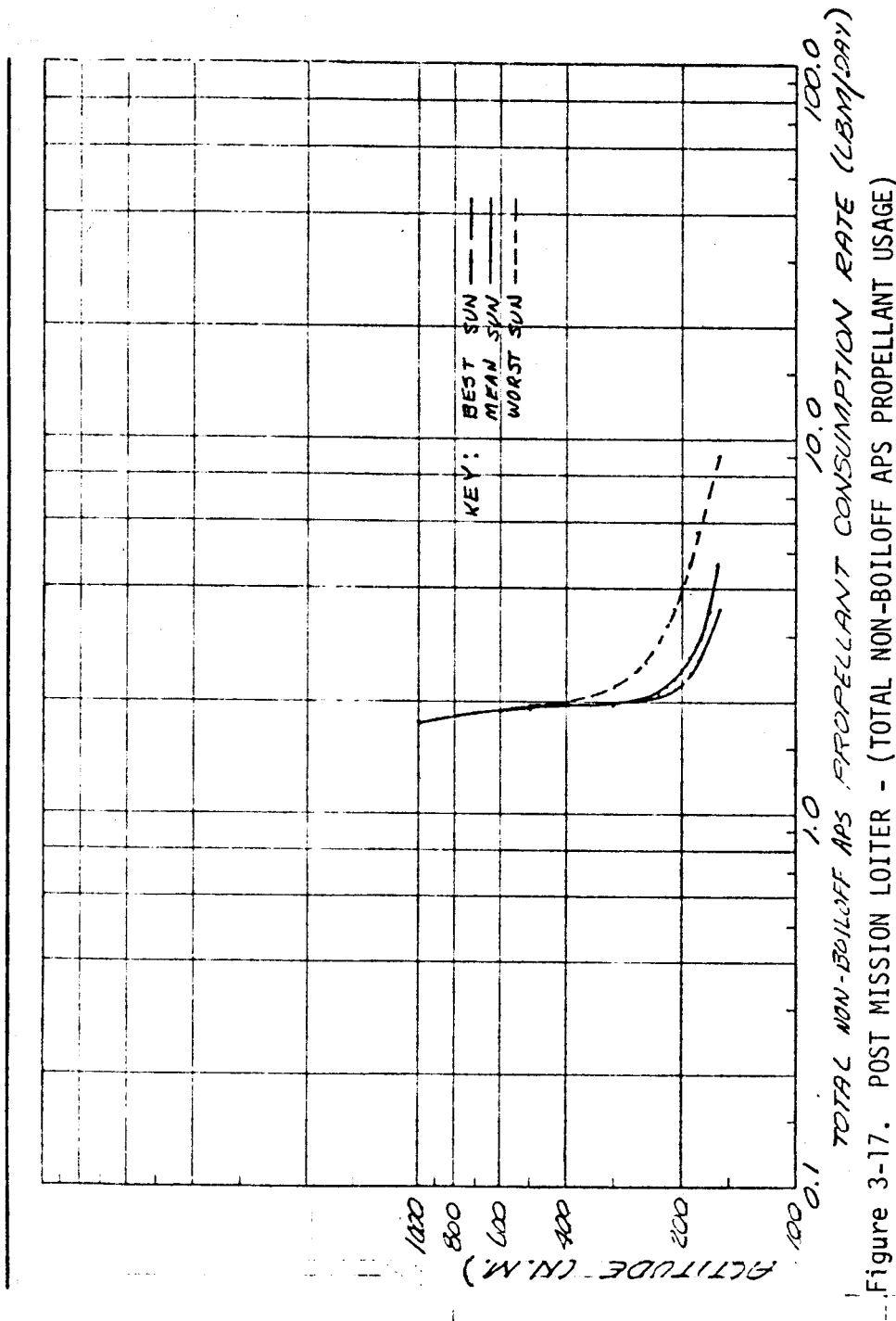


Figure 3-16. PAYLOAD ATTENDANCE LOITER - (TOTAL NON-BOILOFF APS PROPELLANT USAGE)



Deployment, replacement, or servicing of lunar orbiters would be an obvious possibility. It might even be possible to service more than one lunar orbiter on a single mission by optimizing payload versus orbital maneuvering propellant. Another possibility along these lines would be to use the Tug to correct long term changes in the orbits of lunar satellites caused by the disturbing influence of the earth's gravitational field and lunar mass concentrations (mascons).

The lunar orbit payload capability of the Tug is quite sufficient to allow a lunar lander as a payload. The Tug capability to lunar orbit of 10,000 lbm would allow a landed weight of about 6,000 lbm. This should be adequate for an unmanned lunar rover. Another lunar mission which would lend itself well to the extended duration Tug would be a soil-sample-return. In this mission the lander would carry its own descent and ascent propulsion stages. After landing on the lunar surface, the ascent stage would return to orbit carrying a small sample of soil and rendezvous with the waiting Tug. This mission particularly suits the Tug since it will be returning in any case and the additional weight returned would be quite small compared to the weight of the Tug itself. This could provide much needed additional data points in the soil sample survey begun by the Apollo Program at different latitudes and longitudes.

Missions to the Earth-Moon system libration points, L_1 and L_2 also appear to be attractive possibilities for the extended Tug. An isolated two body system has in its gravitational field five points

where a satellite placed with proper velocity will fly in formation with it. At these libration points, as they are frequently called, the centripetal acceleration will exactly balance gravitational ones. Of the five points, two are of particular interest, both of which lie on the Earth-Moon line. The first point, L_1 , lies between the Earth and the Moon, but much closer to the Moon; and second point, L_2 , lies on the far side of the Moon. It is possible to put a satellite in orbit around the libration point L_2 with the plane of the orbit perpendicular to the Earth-Moon line. Such an orbit is referred to as a "halo orbit." If the size of the orbit is chosen correctly it will be just large enough for the satellite to be constantly visible from Earth. Since such a satellite will also be visible from the far side of the Moon, this single satellite can provide a continuous communication link between Earth and the lunar far side. Some such communication link would be necessary for any far side lander type missions. These concepts are visualized in Figure 3-18.

The Tug is capable of deployment, replacement, or servicing such "halo orbit" satellites. It might also be used for refueling since stationkeeping in the halo orbit requires approximately 500 fps per year.

The libration point L_1 on the near side of the Moon also has possibilities for the stationing of a communication satellite. In combination with a satellite in a slightly enlarged halo orbit around L_2 , it could provide a point-to-point communications network covering most of the lunar surface.

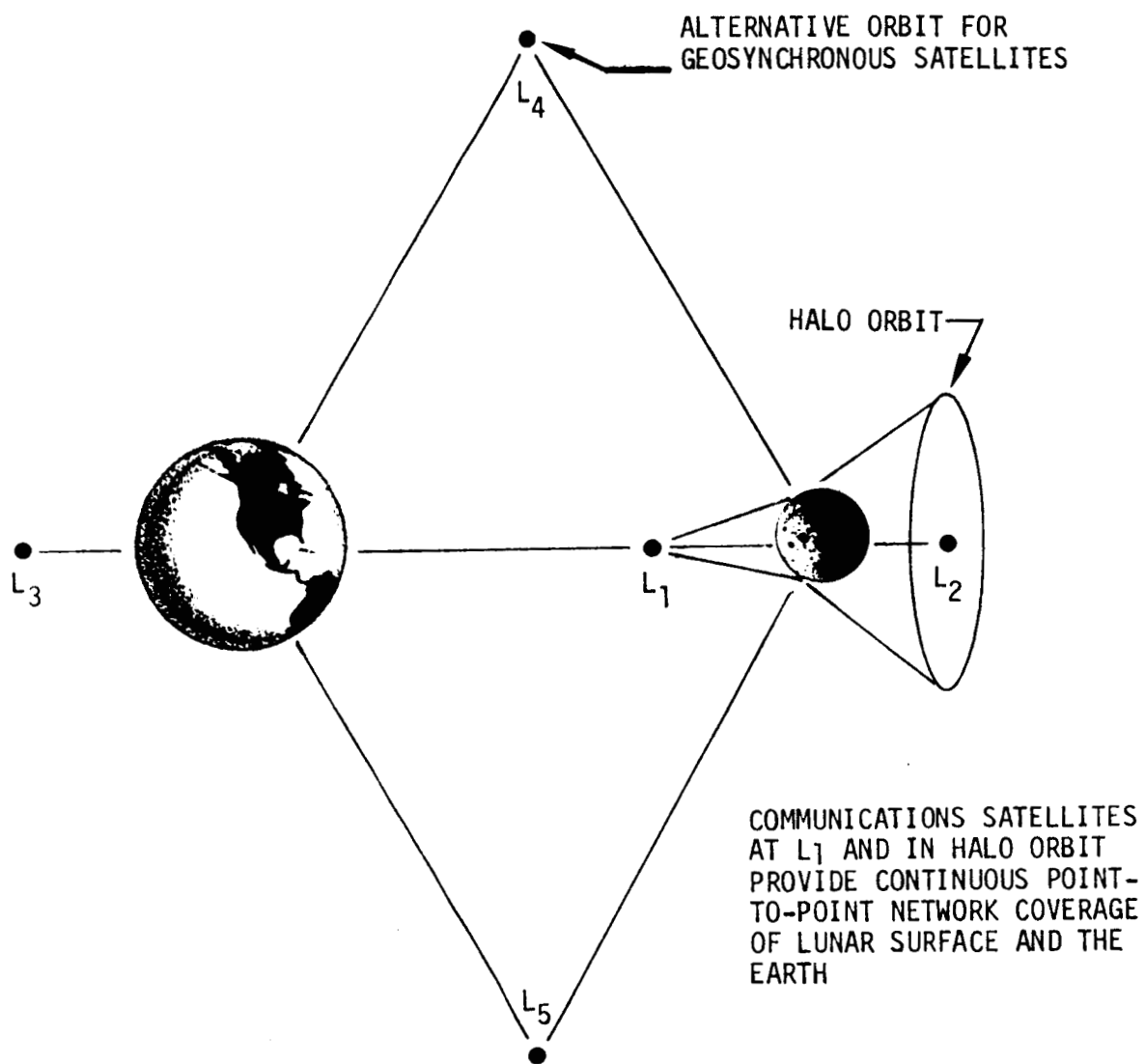


Figure 3-18. SCHEMATIC REPRESENTATION OF LIBRATION POINT MISSIONS

As can be seen from Table 2-1, the Tug has a deployment capability to L_1 in excess of 14,000 lbm. This would allow it to supply a permanent refueling station at L_1 . For higher latitude or longer stay-time lunar missions, transfer via rendezvous with L_1 offers significant advantages.

Lunar and libration point missions appear to be excellent possibilities for an extended Tug. In most respects such missions should not be any more difficult than the ESO and loiter missions presently under consideration. Systems which would require special consideration for these missions include communications equipment and guidance and navigation systems with attendant software.

3.2 THERMAL EFFECTS

An analysis of the thermal effects was performed to determine the feasibility of extending the space tug mission from the presently conceived six days in orbit to longer periods of time. An assumption was made that thermal equilibrium would have been reached in a far shorter period and that thermally conditioned components would be operating in the steady state. This assumption was based upon prior experience with the Saturn IVB cryogenic vehicle. Based on this assumption, the study could concentrate its effects on the major thermal concern namely propellant heating. First, the analysis considered an extended geosynchronous orbit during which the tug operated for up to thirty days; and second, considerations of a low earth orbit during which the tug loitered for up to six months. A detailed discussion of the thermal analysis is given in the following paragraphs.

The thermal analysis was basically conducted in three phases:

- (1) The performance of the tug thermal insulation optimized for a six day geosynchronous orbit for longer duration missions.
- (2) The performance of the tug thermal insulation optimized for longer periods of time handled parametrically.
- (3) The performance of an optional active thermal scheme which offers the potential to drastically reduced propellant losses for longer duration missions up to 6 months. The common factor used to evaluate the results of these three phases of the analysis is weight of propellant lost due to boiloff.

No considerations were given in the first two phases to use the boiloff for other purposes such as attitude control propellants or fuel cell reactants. The third phase did examine the use of the vented gaseous hydrogen to aid in the cooling of the propellants.

The optimized insulation design for the six day tug mission was evaluated for the thirty day geosynchronous orbit. This design consists of 65 layers of multilayer insulation (MLI) at 100 layer pairs per inch for the liquid hydrogen tank and similarly 89 layers of MLI for the liquid oxygen tank. These figures for the number of layers was calculated by solving the heat transfer differential equations for heat losses through the MLI, optimizing the losses and MLI weights and assuming the following constants:

Tank Area (FT^2)

LH_2 - 800

LO_2 - 425

MLI Effective Conductivity ($\text{BTU HR}^{-1} \text{ FT}^{-1} {}^\circ\text{R}^{-1}$)

H_2 Tank - $2.783 \times 10^{-5} - 6.5 \times 10^{-6} \times \text{MLI thickness}$

O_2 Tank - $3.503 \times 10^{-5} - 8.14 \times 10^{-6} \times \text{MLI thickness}$

Heat Shorts (LB DAY^{-1})

H_2 Tank - 5

O_2 Tank - 8

Average Outside Bag Temp (${}^\circ\text{R}$) - 460

The calculations assumed that the propellant in the tanks were complete mixed without stratification. The same equations and constants were used to derive optimum passive thermal protection for other mission periods.

Evaluation was made of the 6 day optimum MLI in order to determine the tug performance made for an extended period in orbit without modifications to the insulation design. As previously noted, the LH_2 and LO_2 tanks were assumed to be continuously vented without stratification in the tanks. It should be noted that the study of the six day tug design consisted of the insulation performance in a quiescent state assuming constant tank pressures and zero -g conditions. However, it was felt that the thermal effects would not differ significantly if intermittent periods of powered flight were factored into the analysis or greatly modify the results obtained.

The second phase of the analysis consisted of thermal assessments of the tug in low earth loitering orbit for extended periods up to six months (180 days). Following the guidelines used in the first phase, the insulation requirements for the LH_2 and LO_2 tank were optimized for several mission durations consisting of a 10 day mission, a thirty day, sixty day, etc., up to six months. For each mission, an optimum insulation system was determined to provide the minimum propellant boil-off and insulation weight penalty.

The third phase examined options or design changes to minimize thermal effects over longer mission periods. The system design discussed above consisted of MLI inserted between an outer shroud and the tank wall. As the mission duration increased, the amount of insulation was increased to prevent excessive propellant losses. Even with the optimum insulation design, it was felt that the added weight penalty due to adding insulation might become prohibitive for the long duration missions. Therefore, it

was felt that an alternate insulation scheme should be investigated which might be somewhat more complicated but provide enough of a weight saving to justify its use.

The design option chosen in the third phase of the analysis employing active cooling loops is shown schematically in Figure 3-19. It consisted of using the hydrogen vent gases to absorb the thermal energy from the shroud. The gases are passed through a heat exchanger consisting of tubes on an 0.005 aluminum shield which is inserted within the insulation blanket. The vented H_2 flows over the LH_2 tank, is heated by the thermal energy from the shroud, and is then passed to a similar heat exchanger (0.005 aluminum shield with flow tubes) which is embedded within the insulation blanket surrounding the LO_2 tank. The gaseous hydrogen which enters the LO_2 tank heat exchanger - MLI sandwich is considerably below the boiling point of LO_2 and when it exits is not much higher than the boiling point of LO_2 . Because of the insulation between the shield and LO_2 tank and the small temperature gradient between the LO_2 can be made to be very small. Consequently, the LO_2 venting losses will be negligible thereby conserving LO_2 . If necessary, the LO_2 pressure can be allowed to increase slightly over long duration missions.

In the third phase analysis three separate designs were considered. The first consisted of having a single aluminum flow shield within the LH_2 insulation blanket and a single shield within the LO_2 insulation blanket. The LH_2 vent would enter the tubes on the aluminum shield on the LH_2 tank at $40^\circ R$ and exit at $60^\circ R$. The vented hydrogen would then pass through

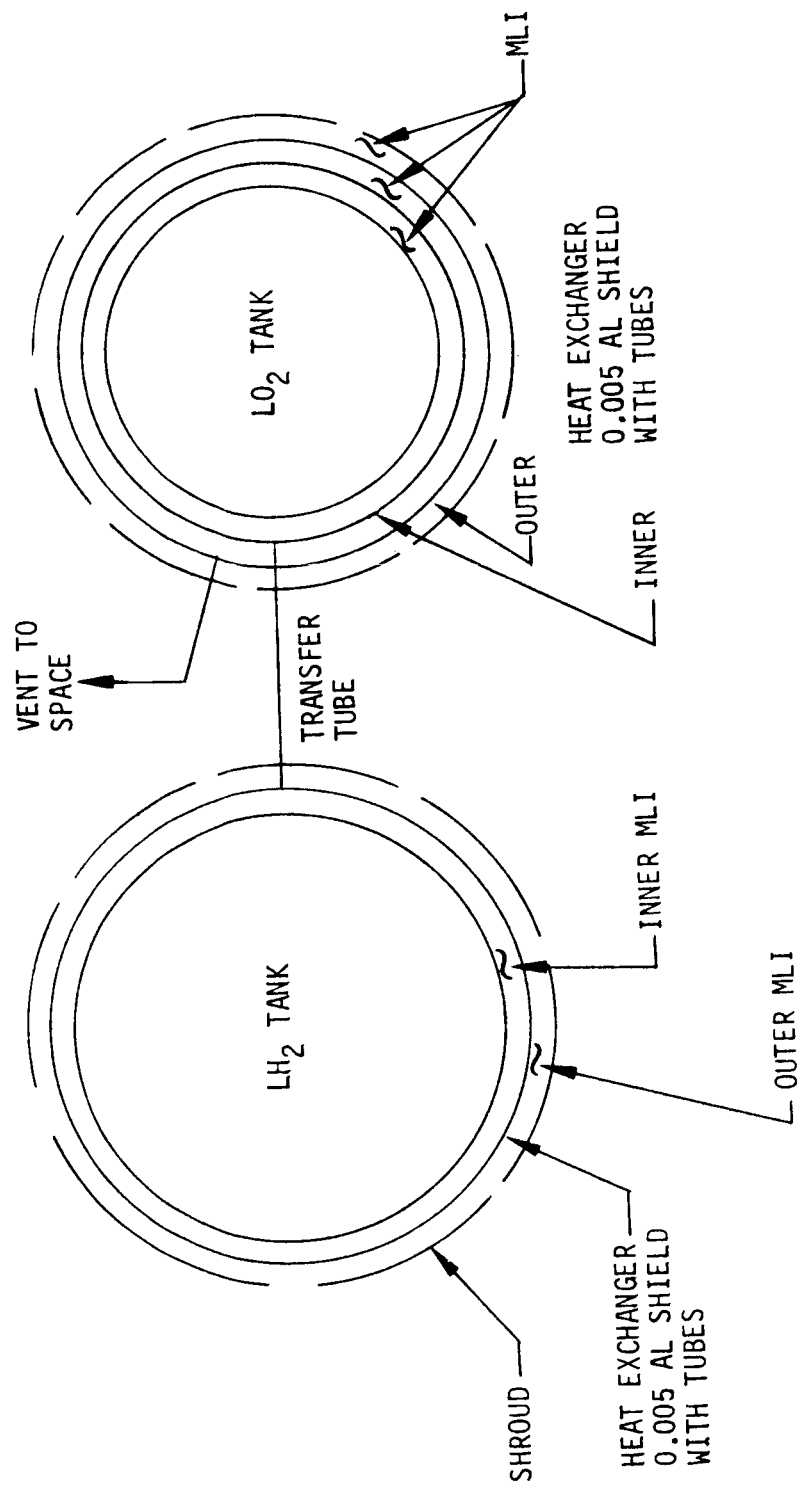


Figure 3-19. SCHEMATIC OF ACTIVE SCHEME FOR PROPELLANT TANK INSULATION AND COOLING

the aluminum shield on the LO_2 tank and exit at 170°R . The study assumed that the vented hydrogen would leave the LH_2 shield at 60°R might be somewhat conservative, so the second design approach was chosen similar to the first except the vented hydrogen was assumed to exit the LH_2 shield at 90°R . The study assumed a third design modification to consider a single shield heat exchanger on the LH_2 tank and a double shield exchanger on the LO_2 tank. As before, the vented hydrogen entered the LH_2 shield at 50°R and exited at 90°R . The vented hydrogen was then passed through the shield adjacent to the LO_2 tank wall. The vented hydrogen exits this shield at 170°R and passes into the shield adjacent to the shroud. The vented hydrogen then exits this shield at 350°R .

The first phase of the analysis, which was an attempt to determine if the six day tug design was satisfactory for a 30 day extended geosynchronous orbit, indicates that a total boil-off of 1312.3 pounds is expected for a 30 day mission. This consists of 698.5 pounds of LH_2 boil-off and 613.8 pounds of LO_2 boil-off. Also, the weight penalty for the MLI consists of 167.1 pounds for the LH_2 tank and 114.1 pounds for the LO_2 tank. This is shown in Figures 3-20. In contrast, an optimized 30 day MLI system consisting of 145 layers of MLI on the LH_2 tank and 199 layers of MLI on the LO_2 tank results in a LH_2 boil-off of 340 pounds of LH_2 and 352.4 pounds of LO_2 . The insulation weights are 327.5 pounds on the LH_2 tank and 352.4 pounds on the LO_2 tank. This is shown in Figure 3-21.

For the 180 day loiter mission which was analyzed in the second phase, an optimum insulation design for a 180 day mission consisted of 355 layers of MLI on the LH_2 tank and 486 layers on the LO_2 tank. This design resulted in a LH_2 boiloff weight of 1019.3 pounds of LH_2 and

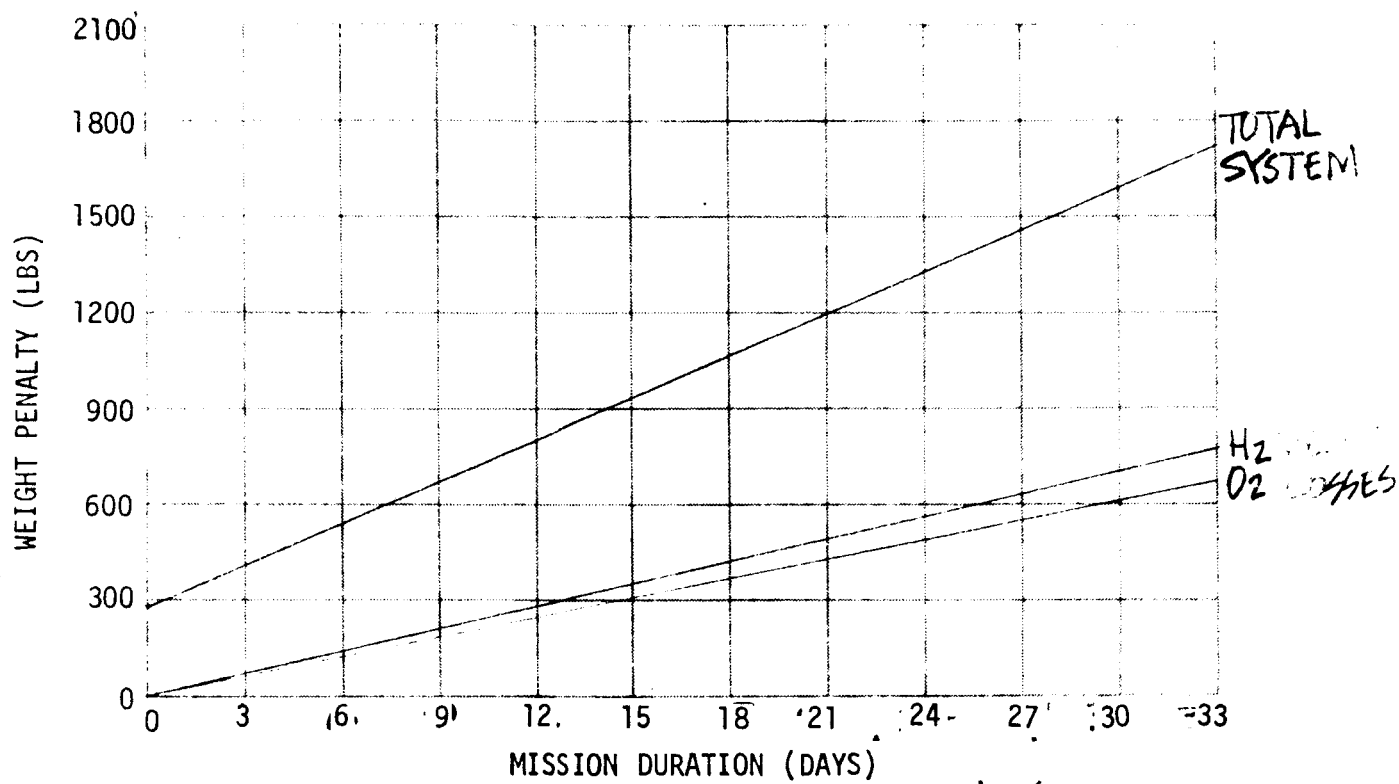


Figure 3-20. MISSION EXTENSION EFFECTS ON 6-DAY OPTIMUM TUG DESIGN THERMAL INSULATION

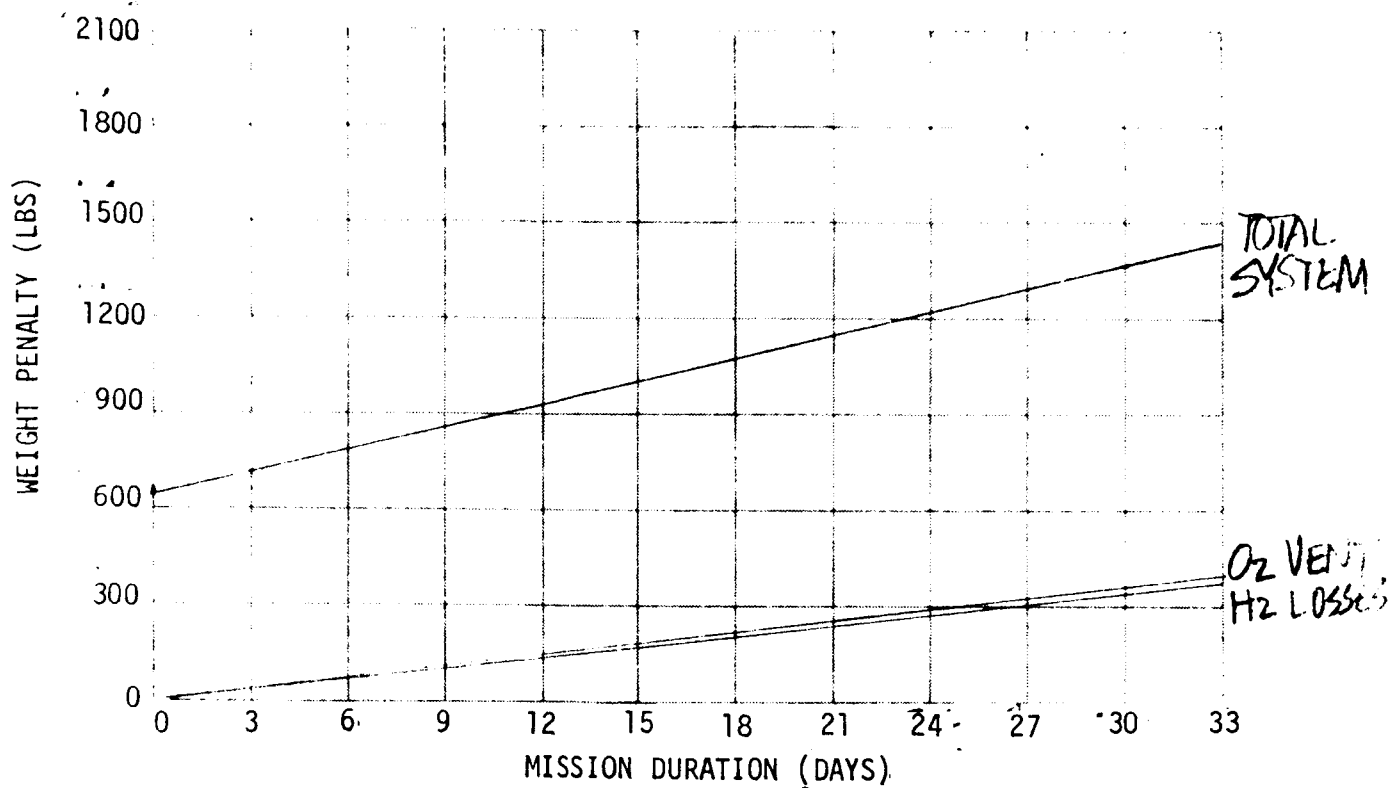


Figure 3-21. MISSION EXTENSION EFFECTS ON 30-DAY OPTIMUM TUG DESIGN THERMAL INSULATION

and 1368.4 pounds of LO_2 for a total weight penalty of 2387.7 pounds of propellant. The weight penalty for the MLI consisted of 747.5 pounds on the LH_2 tank and 535.2 pounds on the LO_2 tank. These results are shown in Figures 3-22 and 3-23. Also shown in Table 3-2 is number of MLI pairs corresponding to various mission durations. It should be noted that Figures 3-22 and 3-23 are used to show optimized designs for any time point on the graph. For example, the optimized boil-off and insulation weight for a 90 day mission will not be optimum for a 120 day mission. The 120 day weight penalty applies only for a 120 day mission.

The results of the third phase analysis is shown in Figure 3-24. It should be noted that this design consisted of the heat exchangers around the LH_2 and LO_2 tank through which vented hydrogen is flowed. For a 180 day mission, if a single shield is on both tanks and the vented hydrogen is exited at 60°R , the LH_2 boiloff weight is 860 pounds and the LH_2 MLI weight 660 pounds, whereas, the LO_2 MLI weighs 280 pounds. The total shield weight for both tanks is 90 pounds. The total weight penalty for this design is 1890 pounds. For the same mission duration, if the vented hydrogen is exited from the LH_2 shield at 90°R , the weight of LH_2 boil-off is 720 pounds and the MLI weight on the LH_2 tank is 580 pounds and the LO_2 tank insulation weight is 390 pounds. The shield weight is the same as for the 60°R exit vented hydrogen case above, e.g., 90 pounds. The total weight penalty for this design is 1780 pounds. For the condition where a single shield is used for a 180 day mission on the LH_2 tank and a double shield is used on the LO_2 tank the weight of LH_2 boiloff is 720 pounds and the LH_2 tank insulation weighs 580 pounds. The LO_2 tank insulation weight is 120 pounds. The shield weight for this system is 130 pounds. The total weight penalty for this design is 1550 pounds.

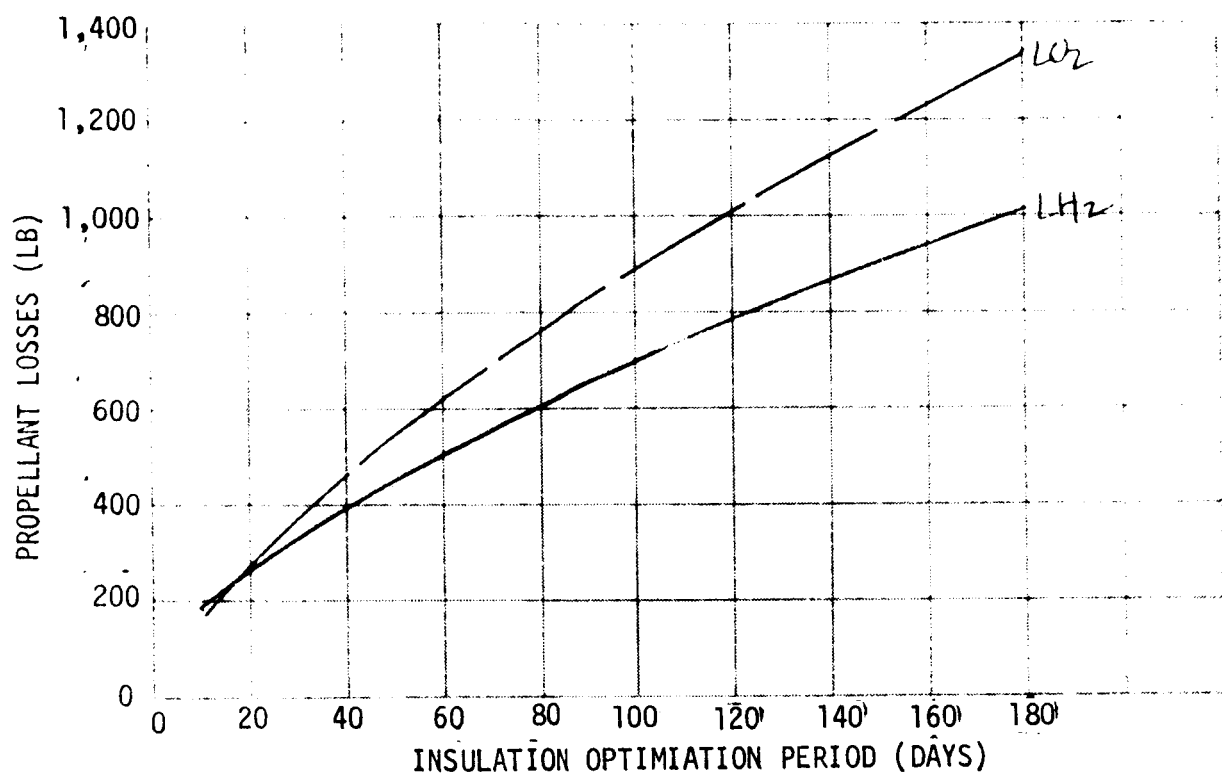


Figure 3-22. PROPELLANT BOILOFF FOR OPTIMIZED MLI

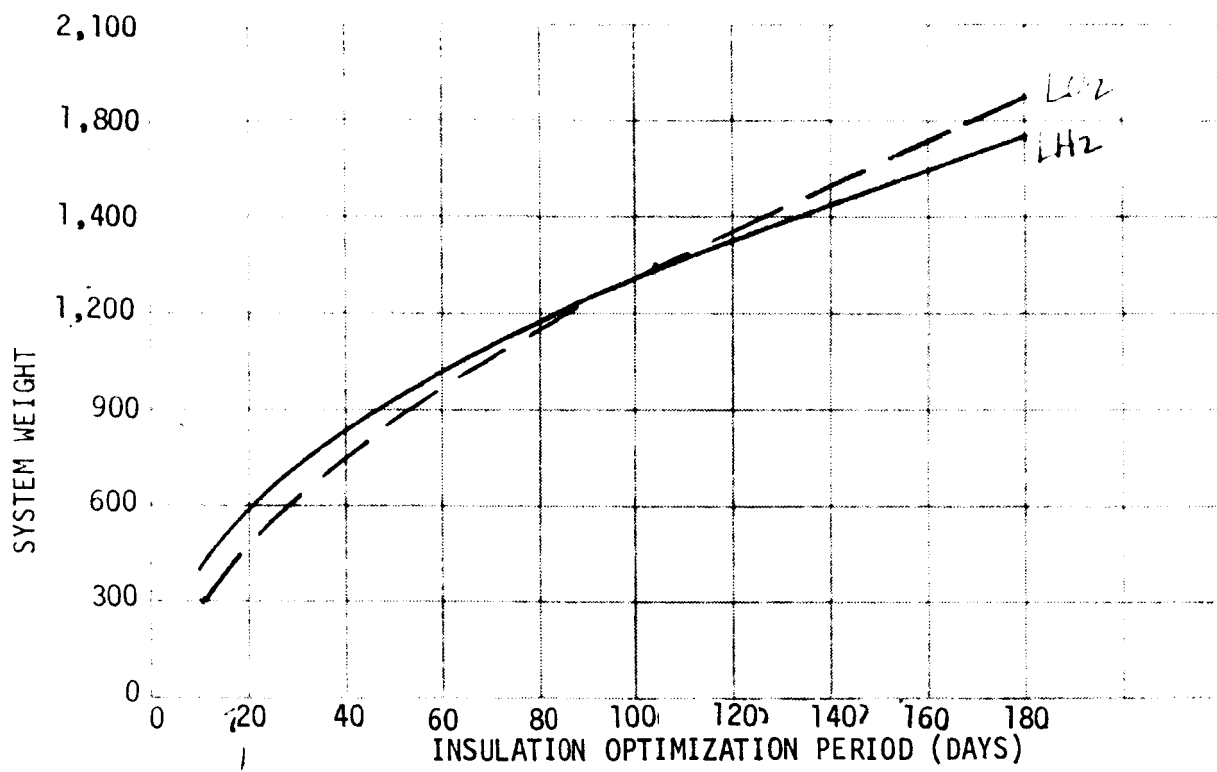


Figure 3-23. OPTIMIZED TOTAL PROPELLANT HEATING REQUIREMENTS

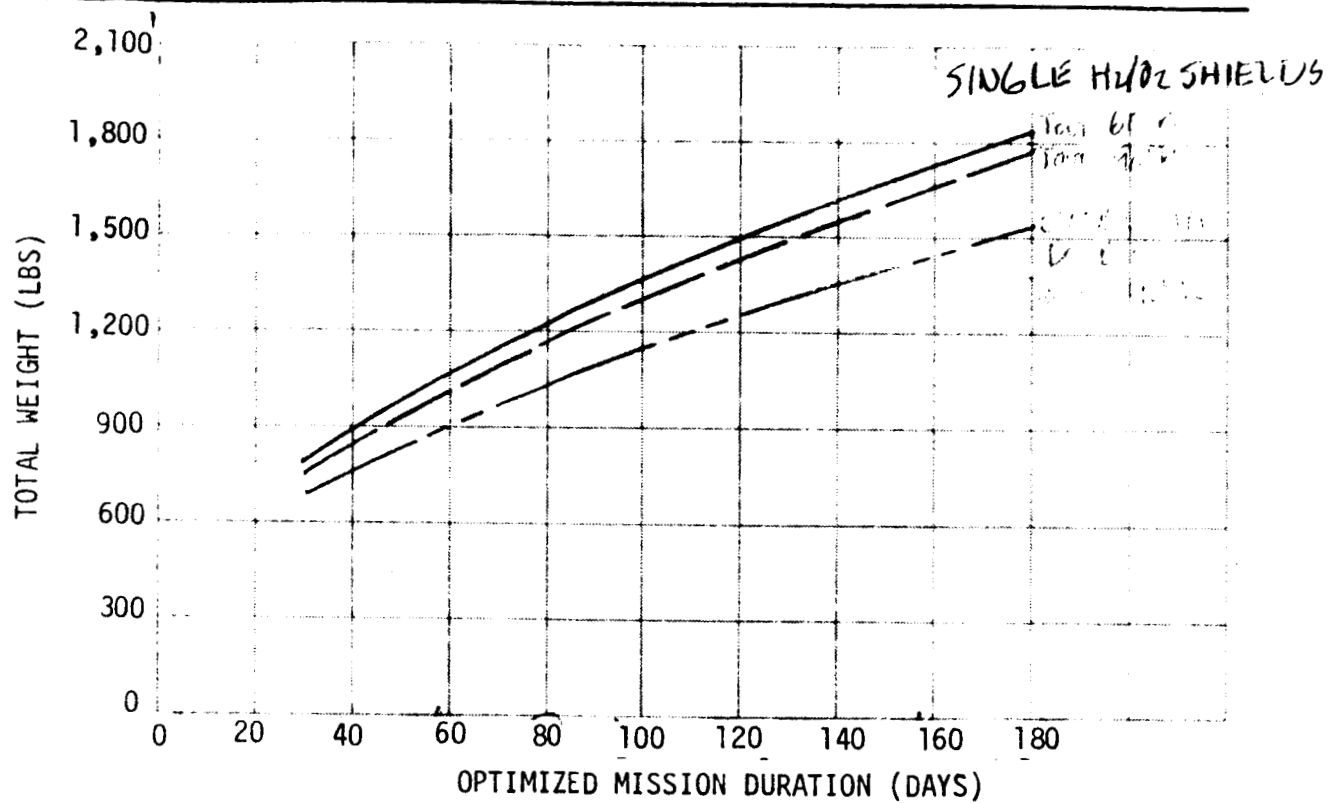


Figure 3-24. ACTIVE COOLING SYSTEM PERFORMANCE

If the present six day tug design is used for an extended 30 day geosynchronous mission, the weight penalty is about 250 pounds compared to the optimized 30 day mission. Although this weight penalty may be tolerable, the optimized design probably would be more efficient and economical for multiple missions. Also, for the 180 day loiter missions, using the LH_2 vent to reduce the LH_2 boiloff and eliminate LO_2 venting results in a minimum weight savings of 1770 pounds and a maximum saving of approximately 2100 pounds over the simple design consisting of MLI on the LH_2 and LO_2 tank alone. It is felt that the shield design would also prove more economical and efficient for multiple missions. By way of summary, Figure 3-25 presents the optimized insulation losses for various mission durations comparing the active and passive systems.

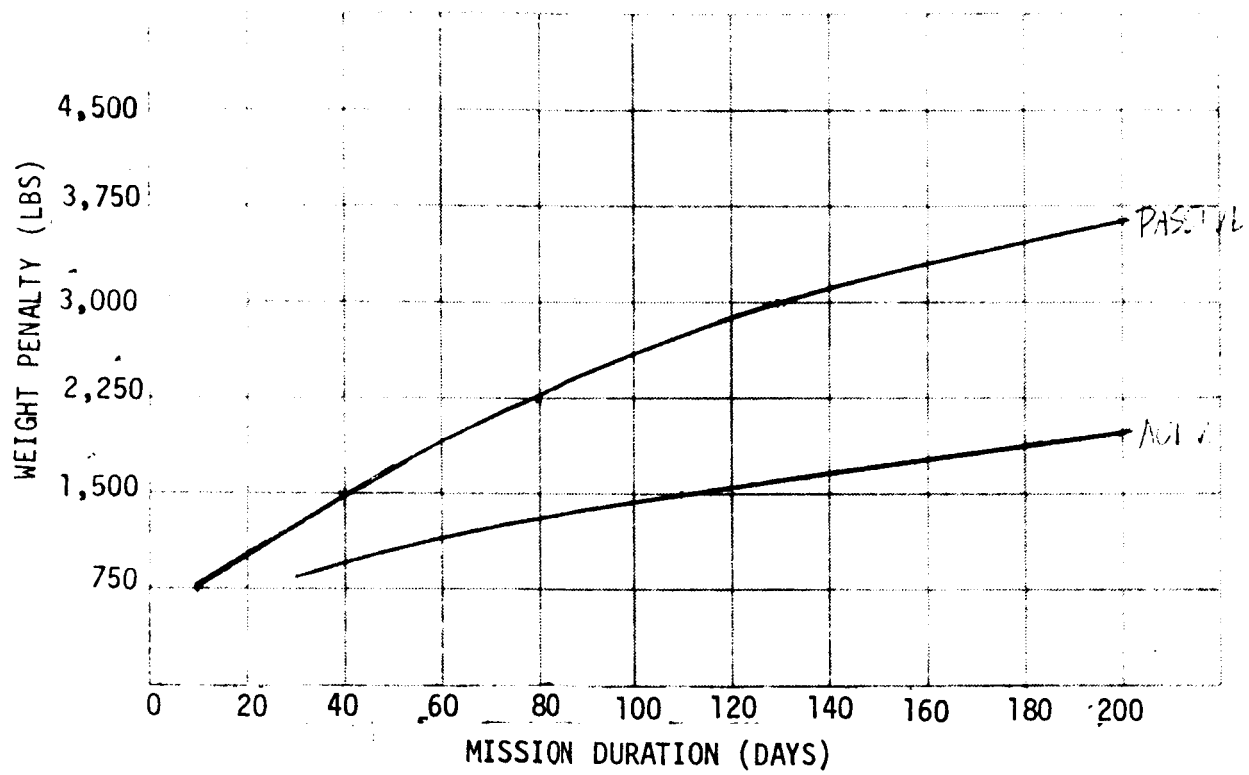


Figure 3-25. OPTIMIZED PROPELLANT HEATING INSULATION PERFORMANCE

3.3 COMBINED FLIGHT EFFECTS

Having available from the study the results of the flight mechanics analysis and the thermal protection consideration, it was possible to combine the data from each respective area and examine comparative factors. Figure 3-26 is a combination plot of (1) a smothered curve derived from

Table 3-1 of propellants required to perform the maneuvers necessary to visit and service four satellites in geosynchronous orbit, (2) the propellant losses and penalties due to heating, and (3) a summation of the total propellant requirements. All 3 curves are plotted to a common mission time parameter. The immediate conclusion that is evident from this plot is that from the standpoint of propellant usage, the most efficient mission times exceed six days and approach 15 days and longer in duration.

Figure 3-27 presents the total propellant penalty associated with "loitering" in orbit. This figure represents the combined effects of many parameters. Data for four different thermal system designs and two different loiter orbit altitudes are included. The term "propellant penalty" is used because the figure not only reflects actual propellant consumed but also increases in stage weight which would require an equivalent amount of propellant to be off-loaded. No attempt has been made to distinguish between LH_2 and LO_2 consumption. It is assumed that simultaneous depletion can be obtained using an active propellant utilization system to vary the engine mixture ratio with negligible effect on I_{sp} .

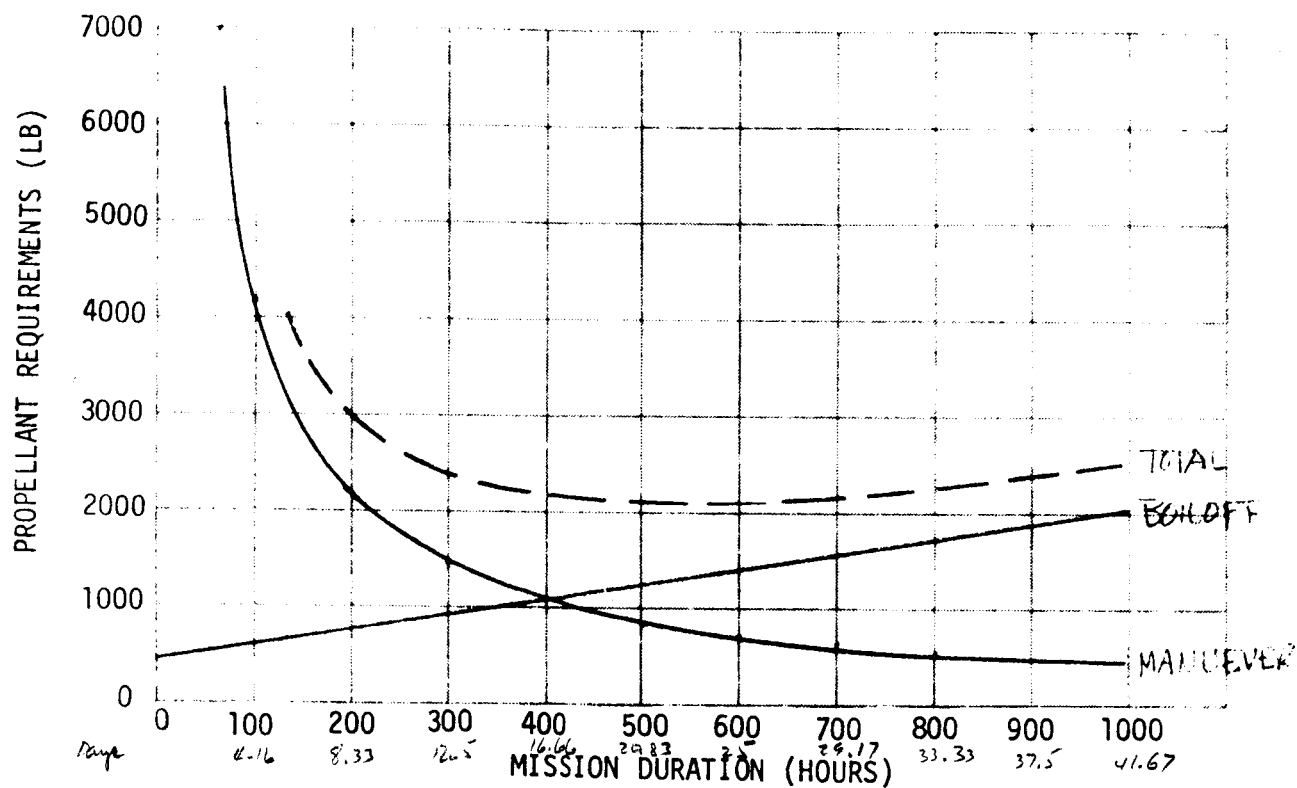


Figure 3-26. 4 SATELLITE SERVICE MISSION PROPELLANT REQUIREMENTS

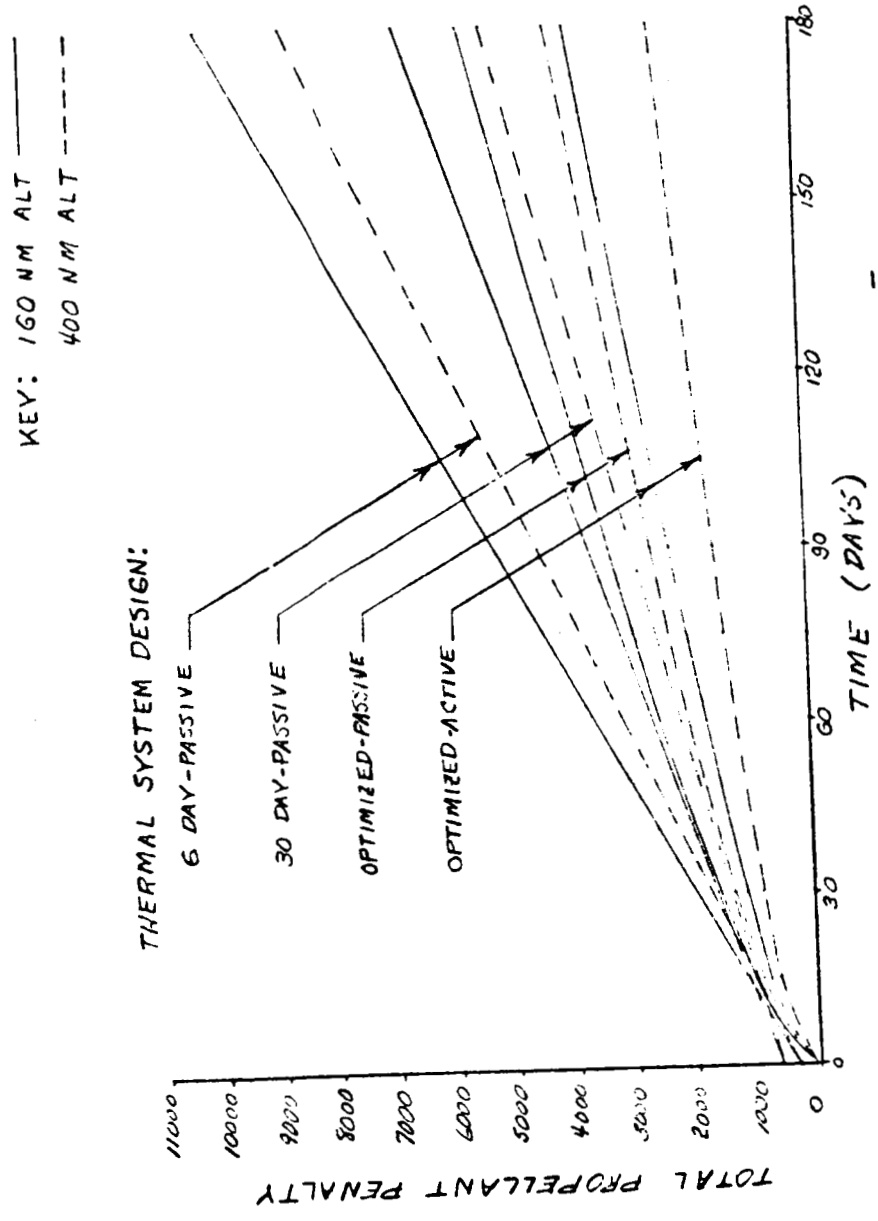


Figure 3-27. TOTAL PROPELLANT PENALTY IN LOITER ORBIT

3-27

Figure 3-27 is not intended to be comprehensive but is intended to show how the data from the thermal section and the flight mechanics section can be combined to obtain a complete picture of loiter orbit propellant consumption. The "total propellant penalty" referred to in the figure includes the effects of attitude control and orbit keeping propellants as well as LH_2 boiloff, LO_2 boiloff, and weight penalties associated with the thermal system used to reduce boiloff.

The attitude control and orbit keeping propellants are based on the payload attendance mission with the "worst sun" assumption. These data were obtained from Figure 3-16. Notice that the data in Figure 3-16 is presented in terms of propellant rates and, therefore, must be multiplied by time before it can be combined with the thermal data. This assumes that the attitude control and orbit keeping propellant consumption rates remain constant over the 180 day mission. Such an assumption is not strictly true since there will be some feedback due to changing mass characteristics. The effect is second order and has been ignored in this preliminary study.

The combined effects of propellant boiloff and thermal system weight penalties were obtained from Figures 3-21, 3-23, and 3-24 for the "six-day passive," "optimized passive," and "optimized-active" thermal systems respectively. The reader is referred to the text accompanying these figures for a complete description of the thermal systems involved. The "six-day-passive" system is briefly a passive thermal system with insulation thickness chosen to minimize the sum of propellant boiloff and

insulation weight for a six-day mission. The "optimized passive" system does the same for the exact mission duration under consideration. This is the system that is redesigned for each mission duration. The "optimized-active" system is an active thermal system redesigned at each time point to minimize the sum of propellant boiloff and system weight. The "30-day-passive" data was obtained from the text of the Thermal section. The text calls out a boiloff of 340 lbm LH₂, 352.4 lbm LO₂ with insulation weights of 327.5 lbm on the LH₂ tank and 230.9 lbm on the LO₂ tank. At a time of zero the thermal weight penalty, will be just the total weight of the insulation, i.e., 558.4 lbm. This penalty will increase at the following rate:

$$\frac{(340 \text{ lbm LH}_2 + 352.4 \text{ lbm LO}_2)}{30 \text{ days}} = 23.08 \text{ lbm/day}$$

Thus the weight penalty for the "30-day-passive" design may be represented by the following equation:

$$\text{Thermal wt penalty} = 558.4 + 23.08t$$

As an example of how Figure 3-27 was generated, assume a mission duration of 90 days and a loiter altitude of 160 n.mi. from Figure 3-16, the total of attitude control and orbit keeping propellants required in a 160 n.mi. orbit assuming "worst sun" is about 11 lbm/day. For the "optimized-passive" thermal system a system weight (boiloff plus insulation) of 1170 lbm LO₂ and 1200 lbm LH₂ may be read from Figure 5 of the Thermal section. The total propellant penalty is then arrived as follows:

$$(1170 \text{ lbm LO}_2 + 1200 \text{ lbm LH}_2) + 11 \text{ lbm/day (90 days)} = 3360 \text{ lbm}$$

For an altitude of 400 n.mi. the total of attitude control and orbit keeping propellants from Figure 3-16 is 2.95 lbm/day. The total propellant penalty is then calculated:

$$(1170 \text{ lbm LO}_2 + 1200 \text{ lbm LH}_2) + 2.95 \text{ lbm/day (90 days)} = 2636 \text{ lbm}$$

This process can be repeated for any desired combination of thermal system, loiter orbit altitude, sun condition, etc.

As previously stated Figure 3-27 is not intended to be comprehensive, but it does illustrate several trends which are worth noting. The most obvious of these is the advantage of optimizing the thermal system for the desired mission duration. At 400 n.mi. the sum of attitude control and orbit keeping propellants has dropped to less than a third of its value at 160 n.mi. Yet the percentage change in the "total propellant penalty" going from 160 to 400 n.mi. is small. This reflects the fact that boiloff propellants dominate attitude control and orbit keeping propellants at low altitudes. At higher altitudes this dominance is even more pronounced.

3.4 SPACE TUG MISSION EXTENSION OPTIONS - METEOROID PROTECTION

In this analysis the study evaluated the current "baseline" space tug meteoroid protection system to determine the maximum exposure duration the configuration can withstand and still meet a .995 probability of no tank failure is defined as both no damage and partial penetration orbital altitude is introduced as a parameter.

Penetration Resistance of Baseline Six-Days Exposure Shielding System

At this time the proposed protection system for the space tug tankage is shown in Figure 3-28. The largest mass M that this configuration can defeat will be computed using two failure criteria:

- (1) No acceptable tank damage.
- (2) Partial tank penetration accepted.

No Acceptable Tank Damage

Reference 2 presents an analytical method for computing required backup sheet thickness as a function of impacting mass, based on the outer sheet efficiency. The outer sheet efficiency is also a function of the meteoroid size (the ratio of shield thickness to particle diameter is the governing parameter), so we are faced with a transcendental problem. We will approach this by first assuming that the shield efficiency is non-optimum, then checking to see if this is true for the computed critical mass.

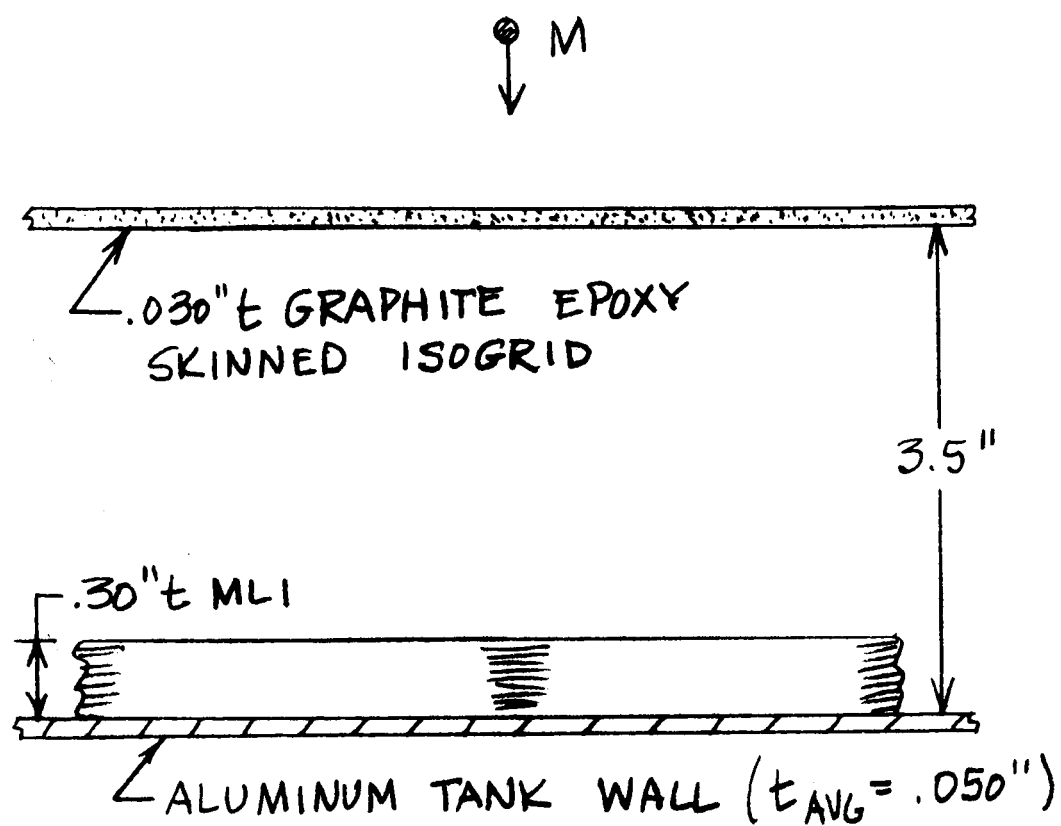


Figure 3-28. METEOROID PROTECTION SHIELD MODEL

The reference 2 method deals with impact on aluminum targets.
Hence, it is necessary first to compute the thickness of aluminum sheet that has the same penetration resistance as the MLI sandwich.

The insulation sandwich is assumed to consist of layers of goldized kapton plus nylon net separators. The ratio of sandwich penetration resistance to that of aluminum is:

$$R = 0.0973$$

Thus, the aluminum sheet thickness equivalent in meteoroid resistance to the .30" MLI is:

$$t_b = (0.0973) (.30) = .02919 \text{ inch}$$

$$t_b = .0741 \text{ CM}$$

From reference 2:

$$t_b = 0.055 (\rho_m \rho_t)^{\frac{1}{6}} m^{\frac{1}{3}} v$$

Where:

t_b = Required Backup Sheet Thickness (CM)

ρ_m = Meteoroid Density (GM/CM³)

ρ_t = Outer Sheet Density (GM/CM³)

m = Meteoroid Mass (GM)

v = Meteoroid Velocity (KM/SEC)

From the meteoroid environment definition model NASA TMX-53957:

$$\rho_m = 0.5 \text{ GM/CM}^3$$

$$v = 20 \text{ KM/SEC}$$

The density of graphite epoxy is:

$$\rho_t = 0.056 \text{ lb/in}^3 = 1.55 \text{ GM/CM}^3$$

Substituting:

$$.0741 = 0.055 [(0.5)(1.55)]^{\frac{1}{6}} m^{\frac{1}{3}} \quad (20)$$

$$m = .705 \times 10^{-1}$$

$$m^{\frac{1}{3}} = 3.5 \times 10^{-4} \text{ GM}$$

This mass was computed using the equation for non-optimum outer shield thickness. We will now check to see if this was valid.

We must first compute the aluminum sphere mass traveling at 7.2 KM/SEC that will cause penetration equivalent to the meteoroid mass:

$$M_m = M_{AL} \left[\frac{\rho_{AL}}{\rho_m} \right]^{.473} \left[\frac{V_{AL}}{V_m} \right]^{2.48}$$

$$3.5 \times 10^{-4} = M_{AL} \left[\frac{2.77}{0.5} \right]^{.473} \left[\frac{7.2}{20} \right]^{2.48}$$

$$M_{AL} = 1.965 \times 10^{-3} \text{ GM}$$

$$\text{Vol} = \frac{M_{AL}}{\rho_{AL}} = \frac{1.965 \times 10^{-3} \text{ GM}}{2.77 \text{ GM/CM}^3}$$

$$\text{Vol} = 7.10 \times 10^{-4} \text{ CM}^3$$

$$\text{Vol} = \frac{\pi R^3}{3}$$

$$R^3 = \frac{3}{\pi} (7.10 \times 10^{-4} \text{ CM}^3)$$

$$R^3 = 1.695 \times 10^{-4} \text{ CM}^3$$

$$R = .0554 \text{ CM}$$

$$\text{DIA } d = .1108 \text{ CM} = .0437 \text{ inch}$$

We must now convert the graphite epoxy outer shield (bumper) to an equivalent aluminum bumper. Reference 5 states that bumper material efficiency is directly related to material density. Thus:

$$ts_{AL} = ts_{G.E.} \left(\frac{\rho_{G.E.}}{\rho_{AL}} \right)$$

$$ts_{AL} = .030 \left(\frac{.056}{.10} \right)$$

$$ts_{AL} = 0.0168 \text{ inch}$$

$$\frac{ts_{AL}}{d} = \frac{.0168}{.0437} = 0.38$$

The non-optimum range begins at $\frac{ts}{d} = .30$ so that the equation used is valid.

The probability of not being impacted by this mass is given by:

$$P(x=0) = e^{-NAT}$$

Where:

N = Particles of mass M or greater per square foot per day

A = Exposed Surface Area (FT²)

T = Exposure Duration (Days)

The total flux of meteoroids (N) consists of stream and sporadic meteoroids. While the stream meteoroid flux is independent of orbital altitude, the sporadic flux does depend on orbit, as shown below.

For 235 N. mile orbit above earth (Skylab)

$$N_1 = -10.594 - 1.22 \log M$$

For an orbital altitude of 13100 N. miles

$$N_1 = -10.692 - 1.22 \log M$$

For a synchronous orbit of 19300 N. miles

$$N_1 = -10.706 - 1.22 \log M$$

The flux of stream meteoroids is constant at an average annual rate of:

$$N_2 = -11.504 - \log M$$

The exposed surface area of the space tug has been previously computed to be:

$$A = 1228.5 \text{ FT}^2$$

It is required that the probability of no failure due to impact meets 0.995. Thus, the only unknown is the exposure duration that the space tug can withstand and still meet the requirements.

Probabilities of no failure as a function of exposure duration and orbital altitude were computed and plotted as shown on Figure 3-29.

Partial Tank Penetration

Using a threshold spallation failure criterion, it has been calculated that the portion of tank thickness that can be considered effective in resisting impact is:

$$t_{EF} = 0.125 \text{ inch} = .0317 \text{ CM}$$

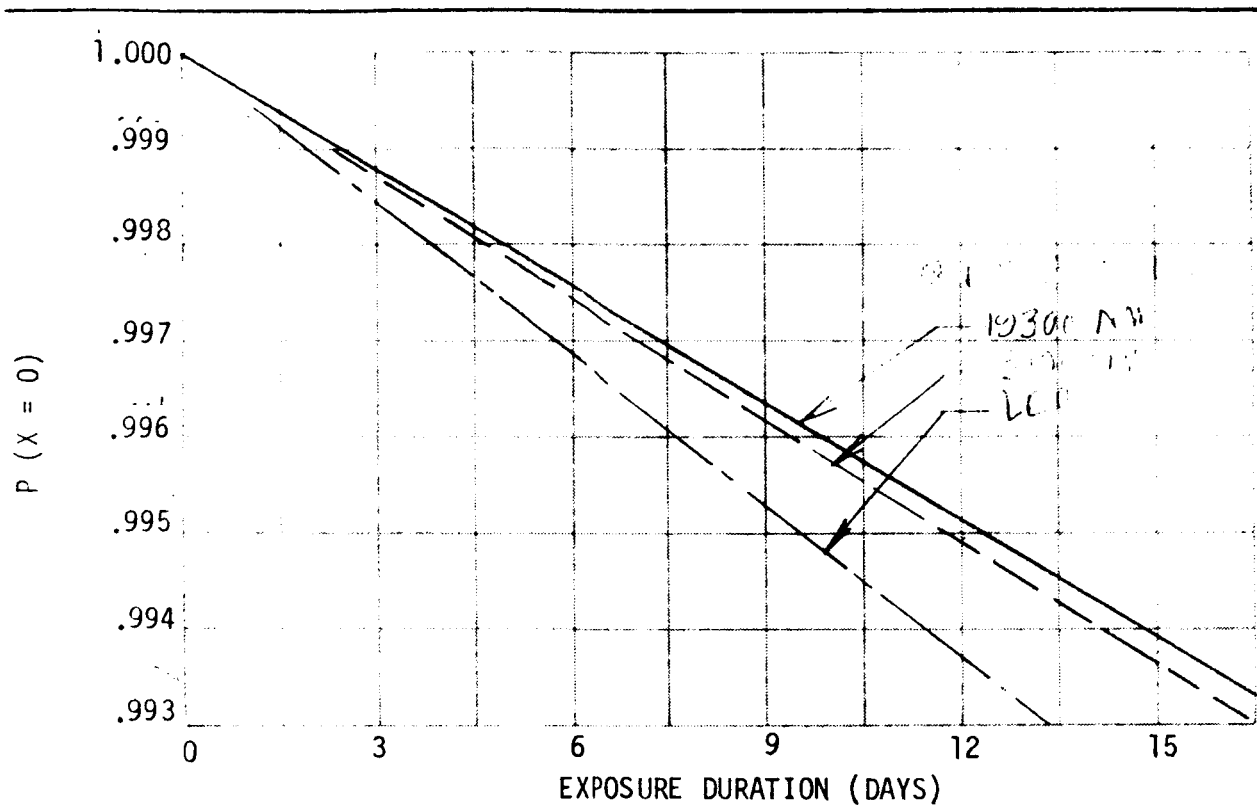


Figure 3-29. BASLINE TUG - TANKAGE NON-EFFECTING

Thus, the total equivalent backup sheet thickness is:

$$t_b = .0317 + .0741 = .1058 \text{ CM}$$

From reference 2:

$$t_b = 0.055 (\rho_m \rho_t)^{\frac{1}{6}} m^{\frac{1}{3}} v$$

$$.1058 = 0.055 [(0.5)(1.55)]^{\frac{1}{6}} m^{\frac{1}{3}} (20)$$

$$m^{\frac{1}{3}} = 1.007 \times 10^{-1}$$

$$m = 1.03 \times 10^{-3} \text{ GM}$$

We now compute the equivalent aluminum sphere mass traveling at 7.2 KM/SEC.

$$M_M = M_{AL} \left[\frac{\rho_{AL}}{\rho_m} \right]^{.473} \left[\frac{V_{AL}}{V_m} \right]^{2.48}$$

$$1.03 \times 10^{-3} = M_{AL} \left[\frac{2.77}{0.5} \right]^{.473} \left[\frac{7.2}{20} \right]^{2.48}$$

$$M_{AL} = 5.78 \times 10^{-3} \text{ GM}$$

$$Vol = \frac{M_{AL}}{\rho_{AL}} = \frac{5.78 \times 10^{-3} \text{ GM}}{2.77 \text{ GM/CM}^3}$$

$$Vol = 2.09 \times 10^{-3} \text{ CM}^3$$

$$Vol = \frac{4}{3} \pi R^3$$

$$R^3 = \frac{3}{4\pi} (2.09 \times 10^{-3})$$

$$R^3 = 4.97 \times 10^{-4} \text{ CM}^3$$

$$R = .792 \text{ CM}$$

$$\text{DIA } d = .1584 \text{ CM} = .0624 \text{ inch}$$

$$\frac{t_s}{d} = \frac{.0168}{.0624} = 0.27$$

This value is in the transition range between optimum and non-optimum equation domains, so the use of the non-optimum equation is conservative.

Probabilities of no failure as a function of exposure duration and orbital altitude were again computed and plotted in Figure 3-30.

Configuration Improvements for Mission Extensions

We have evaluated the baseline configuration and shown that it meets the 0.995 probability of no failure for mission durations of 9-13 days with no acceptable tank damage and 35-45 days with partial tank penetration. In this section we will examine what configuration changes must be made to extend these exposure times.

We first must compute the meteoroid masses that we have to defeat in order to attain the 0.995 reliability level for a particular mission duration. This is accomplished for each of the reference altitudes and the result plotted in Figures 3-31 and 3-32.

The general barrier concept of a skinned isogrid outer wall made from graphite epoxy ($t_{\min} = 0.030"$) spaced off of an insulation sandwich covering the tankage is highly desirable from structural, weight, and insulation standpoints. Hence, we will still address

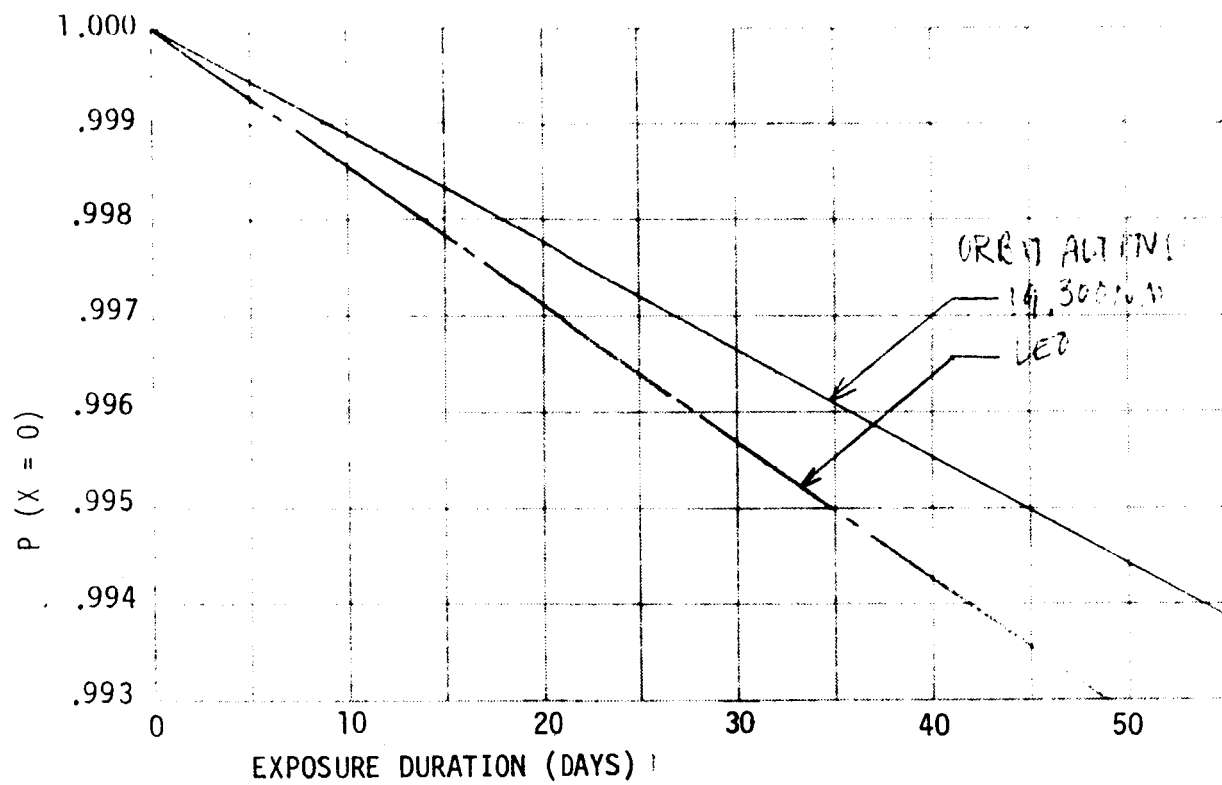


Figure 3-30. BASELINE TUG - EFFECTIVE TANKAGE

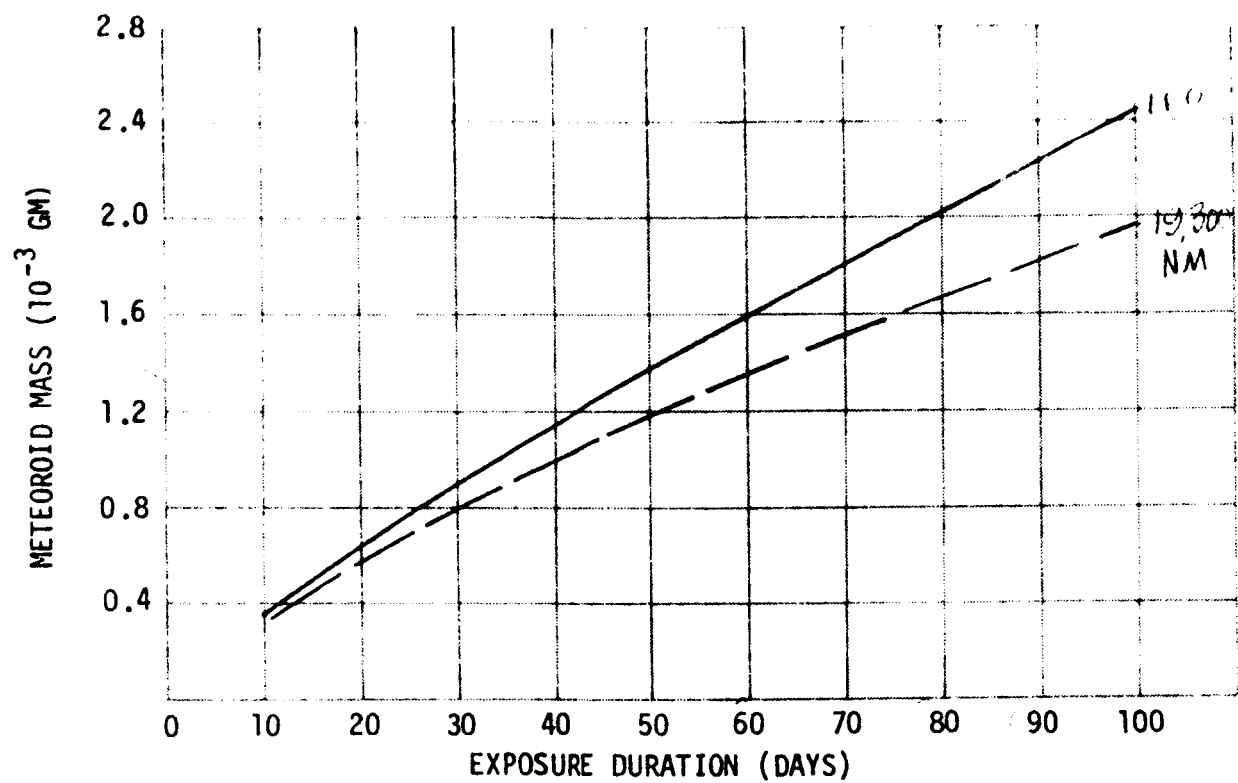


Figure 3-31. CRITICAL METEOROID MASS FOR $P(X = 0) = 0.995$

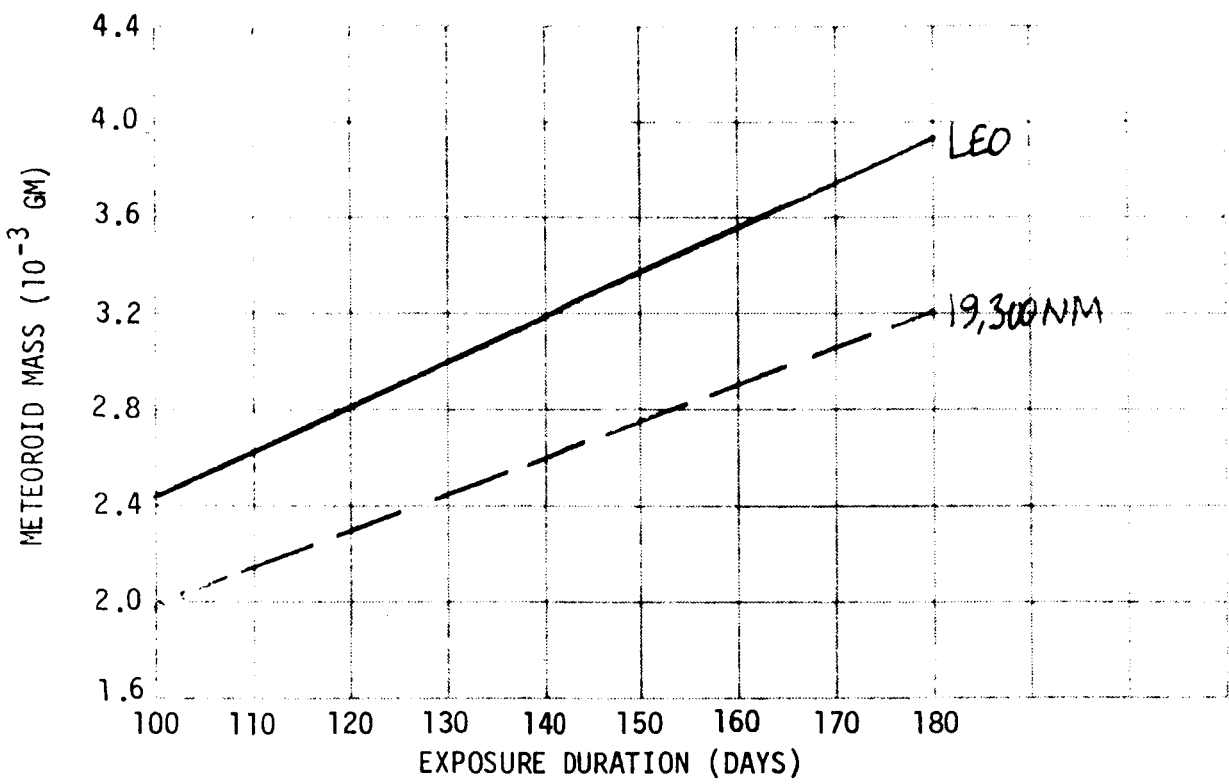


Figure 3-32. CRITICAL METEOROID MASS FOR P (X = 0) - 0.995

this basic configuration, this time looking at the component parameters. These are:

1. Bumper Thickness
2. MLI Thickness
3. Spacing between Bumper and MLI

Again we will consider both no tank damage and partial tank penetration failure criteria.

No Tank Damage

Of the three parameters listed above, increasing the MLI thickness as mission duration increases is the only certainty based on requirements other than meteoroid. Hence, we will use this as the primary parameter in our "design" process. The logic for this process is:

1. Determine required MLI thickness based on the equation on page
2. Compute spacing between graphite epoxy and MLI.
3. Compare this spacing to optimum value of $30d$, where d is the diameter of the equivalent aluminum sphere mass traveling at 7.2 KM/SEC.
4. Compute Ts/d ratio to evaluate bumper efficiency.

The process is performed computing required values as a function of meteoroid mass.

The maximum critical mass for a six month mission (orbital altitude = 19300 N.M.) is 4.0×10^{-3} GM. The spacing between graphite epoxy and MLI for this mass case is 29D. This is still essentially optimum. The bumper efficiency up to this mass size is excellent. Thus, this appears to be the best way of increasing meteoroid protection.

A plot of required MLI t vs exposure duration is made on Figure 3-33.

Partial Tank Penetration

The same approach will be used here as was used in the no tank damage case. Program TMLI will be modified to reflect the added protection provided by the tank wall ($t_{EP} = .0317$ CM) by:

$$170 \quad T2 = (T - .0317) / (2.54 * 0.0973)$$

Results are shown in Figure 3-34.

When comparisons are made between the requirements for MLI in terms of thermal protection and for meteoroid damage protection it is evident that the thermal consideration prevail. Therefore for extended missions, adequate thermal protection for a six-day meteoroid protection for missions extending to six months duration.

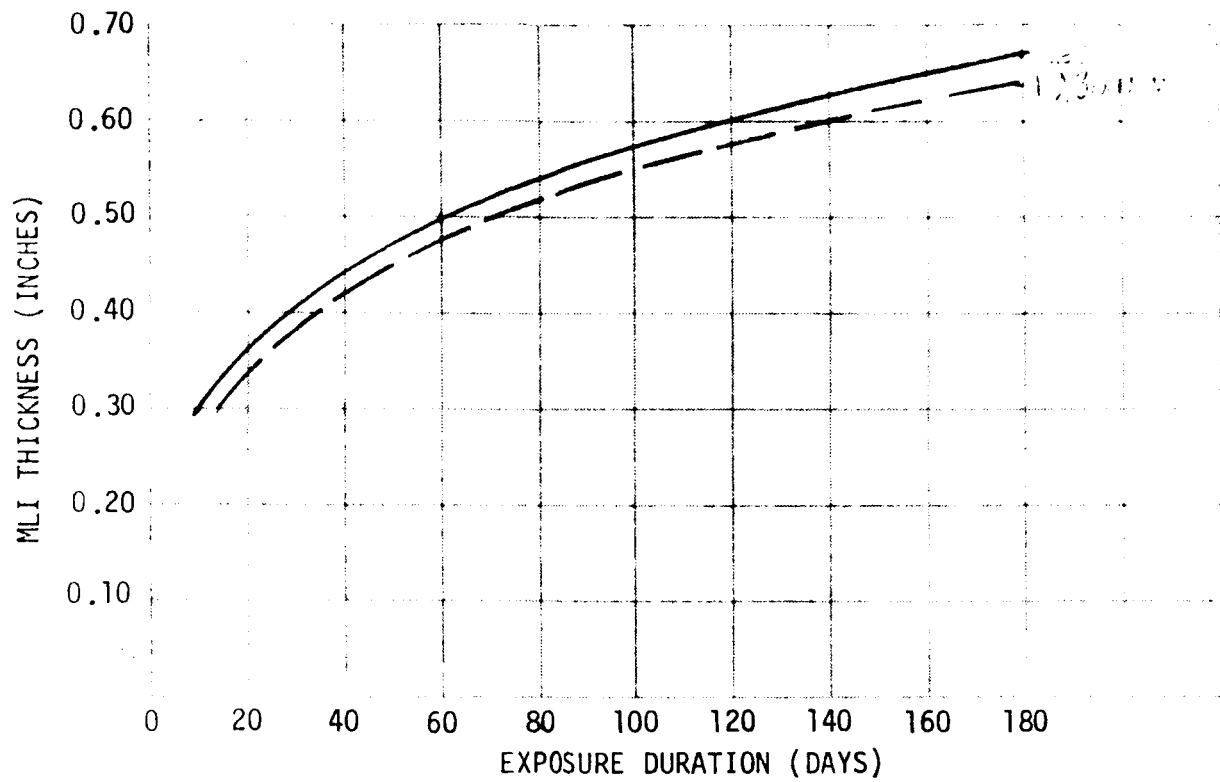


Figure 3-33. MLI REQUIREMENT TO MEET $P(X = 0) = 0.995$ - NONEFFECTIVE CASE

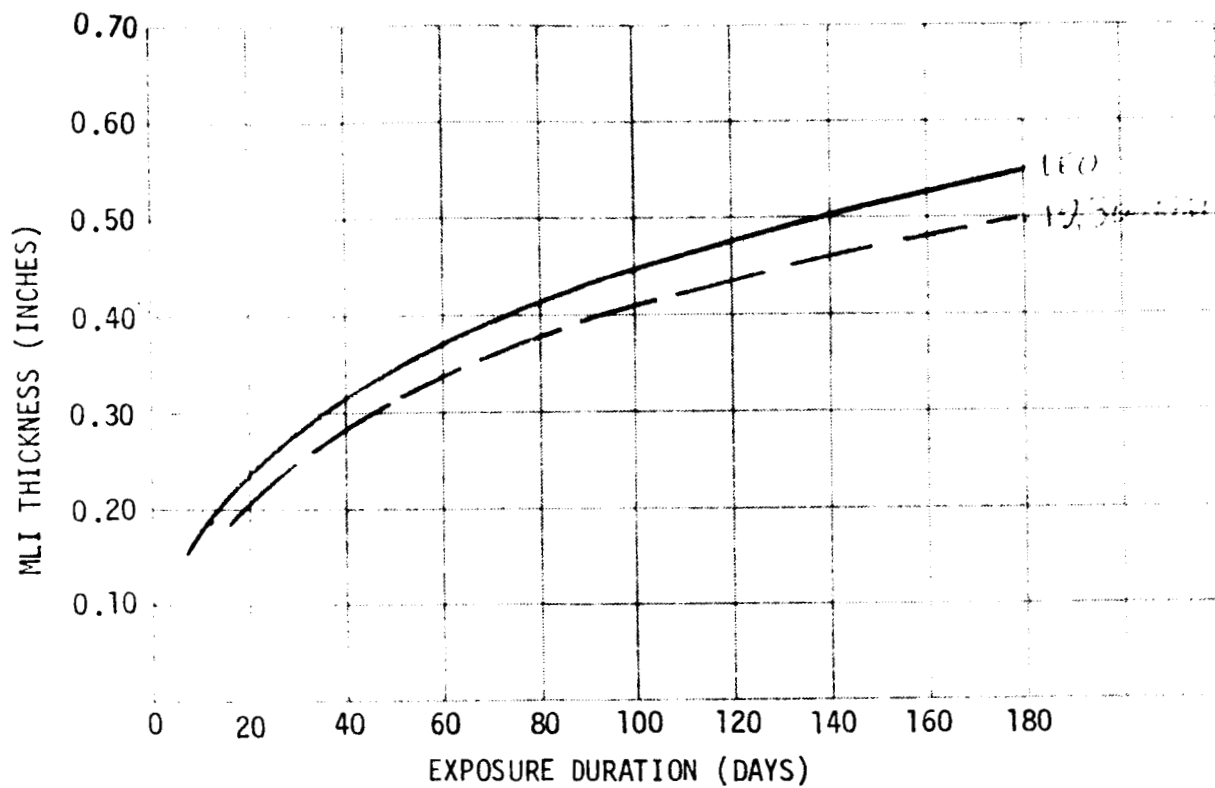


Figure 3-34. MLI REQUIREMENT TO MEET $P(X = 0) = 0.995$ -EFFECTIVE CASE

References

1. "Meteoroid Penetration Resistance of Space Tug Systems Study (Cryogenic) Configurations," Technical Memorandum A3-250-ABCA-TM-26, July 1973.
2. Cour-Palais, B. G., "Meteoroid Protection by Multi-Wall Structures," AIAA Paper 69-372, April 1969.
3. Weidner, D. K., et.al., "Space Environment Criteria Guidelines for Use in Space Vehicle Development (1969 Revision)," NASA TMX-53957, October 1969.
4. "Meteoroid Damage Assessment," NASA SP-8042, May 1970.
5. Swift, H. F. and Hopkins, A. K., "The Effects of Bumper Material Properties on the Operation of Spaced Hypervelocity Particle Shields," AIAA Paper 69-379, April 1969.
6. Orbital Workshop Strength Analysis Report, Volume III (Supporting Studies), Technical Memorandum A3-250-ABC8-TM-3, July 1972.
7. "Space Tug Point Design Study Strength Analysis," Technical Memorandum A3-250-ABCA-TM-8, July 1972.
8. "Meteoroid Protection of Insulation-Protected Space Tug," Memorandum A3-250-ABCA-M-17, October 1972.
9. Arene, R. J., "Influence of Hypervelocity Projectile Size and Density on the Ballistic Limit of Dual-Sheet Structures," AIAA Paper 69-376, April 1969.

3.5 EFFECTS OF MISSION DURATION EXTENSION ON ELECTRICAL POWER SYSTEM

Extension of Space Tug's operating mission duration beyond the six days currently envisioned will require additional reactants to furnish chemical energy for conversion to electrical energy for the vehicle loads. Fuel cells currently ready for program development have been tested for 5000 hours of operation with no discernable performance degradation. The advanced fuel cells being considered for the full capability Tug, for example the 402 or 403 options, will probably exhibit operating lifetimes well in excess of this value.

The various candidate Tug programs will require average power during coast operations between 550 and 870 watts according to present estimates.

Configuration option 403 is estimated to require an average power level of 833 watts. The advanced fuel cell batteries that will supply electrical power for this program are expected to consume oxygen and hydrogen reactants at a combined rate of 0.73 lb/hr, in an 8:1 mass ratio.

The Tug can be operated in one of two ways for an extended coast period. In the first approach, all electrical loads can be operating during the extended coast period that would normally be operating while coasting during a 6 day mission. In the second approach, all loads but the essential ones (including uplink communications gear and the portion of the Data Management System required to process uplink commands) can be turned off. Heaters would have to be added to keep the remaining equipment from getting cold enough to compromise its reliability for subsequent operations. This approach takes into account that thermal cycling can be a major contributor to equipment failures.

It is difficult at the present time to credibly estimate how much reduction in average power requirements could be accomplished by using the powered down approach. Thermal parameters of the many electrical loads vary widely, and detailed analysis will be required for each item of equipment to determine its individual heating requirements.

The total reactant requirement for a program 403 Tug vehicle operating according to the first approach would be 530 lb. of oxygen and hydrogen in a 8:1 mass ratio for a 30 day coast period, or 3200 lbs. for a 6 month coast period. The reactant requirements for the second approach would be undetermined lesser quantities.

$$\frac{W_D}{W_F}$$

$$W_P = W_D - W_F$$

$$\frac{W_P}{W_F} = 1.72$$

$$W_F = \frac{W_P}{1.72} = \frac{530}{1.72} = 308.19 \text{ lb. } \text{GH}_2$$

$$W_D = \frac{W_P}{1.72} = \frac{530}{1.72} = 308.19 \text{ lb. } \text{O}_2$$

355.50
2844.44
<hr/>
3200.00

3.6 SYSTEM RELIABILITY CONSIDERATIONS

In order to achieve the required confidence of mission success reliability considerations must take into account numerous facets of the Tug design. The space tug needs to survive the manufacturing process, transportation to the launch site, checkout and mating with the payload spacecraft, loading into the space shuttle payload bay, launch pad loading and checkout, liftoff and atmospheric powered flight, orbital insertion, deployment from the shuttle bay, the tug mission itself, orbital recovery to the shuttle bay, entry and earth return, transportation to the refurbish area, maintenance and rework to the flight readiness state and the repeat of the process many times in the conduct of subsequent missions. At each stage of the sequence failures which would adversely affect mission success can be encountered or for that matter induced by faulty procedures. It is necessary to assess and examine each facet in order to determine the suitability of the design to survive an extension in space tug mission duration within the larger framework of total system performance. Failures which would compromise mission success can be categorized in three basic categories, namely: (1) defects in materials and workmanship which are normally reduced to an acceptable minimum by inspection, test and checkout procedures during the manufacturing process and prior to initiation of the mission, (2) fatigue failures caused by wearout of components with operating time and duty cycles which are controlled by selection of appropriate long life items or by switchover to redundant elements, and (3) catastrophic failures which are minimized by establishing acceptable safety factors and design margins.

At this stage of the study a preliminary reliability analysis of the effects on system reliability of extending the mission time was made. Configuration option 403, which is very similar in a reliability sense, was used as a model. Figure 3-35 is a projection of the 403 configuration to a mission duration of 30 days. The data depicted shows an estimate of mission completion reliability of about 92 percent for this extended mission period.

Also shown on Figure 3-35 are estimates of the reliability projections for the propulsion system and for the avionics at both high and low levels of operation. Figure 3-36 substantiates the propulsion estimate in terms of a standard series system statistical estimate. A similar approach was used in the avionic computations.

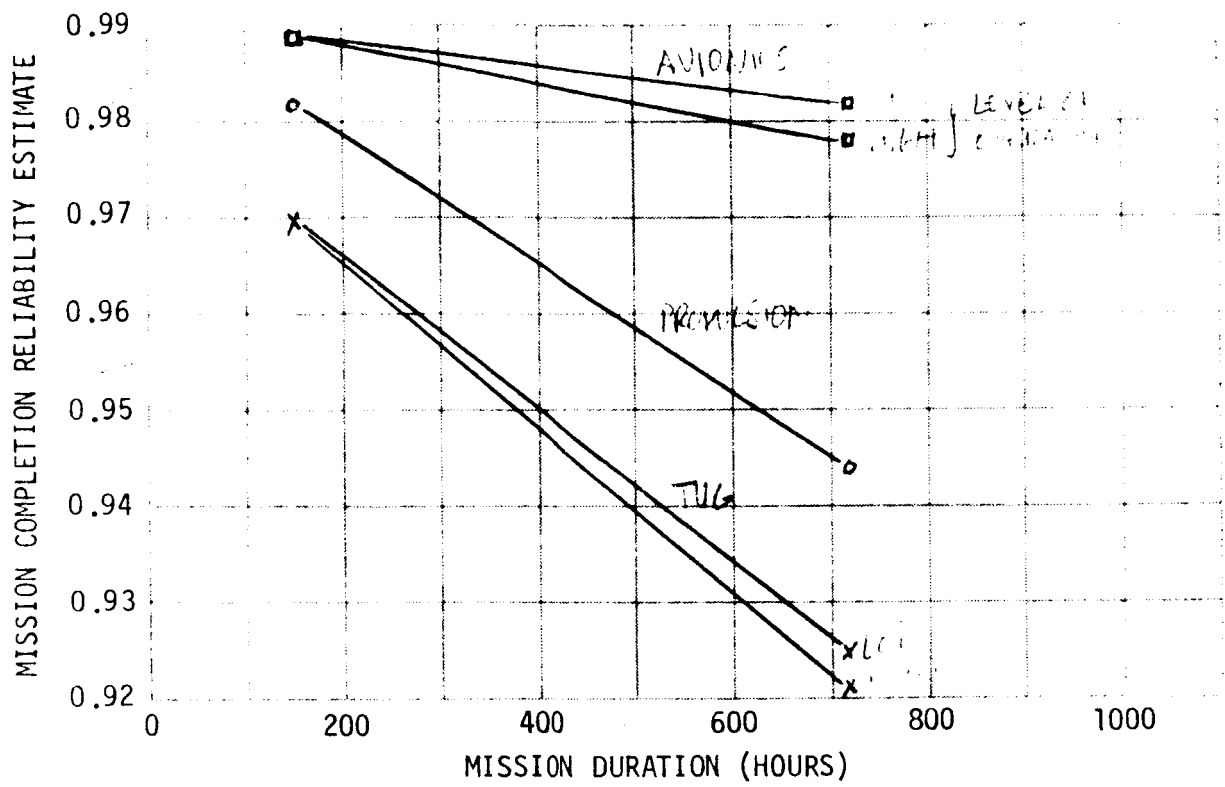
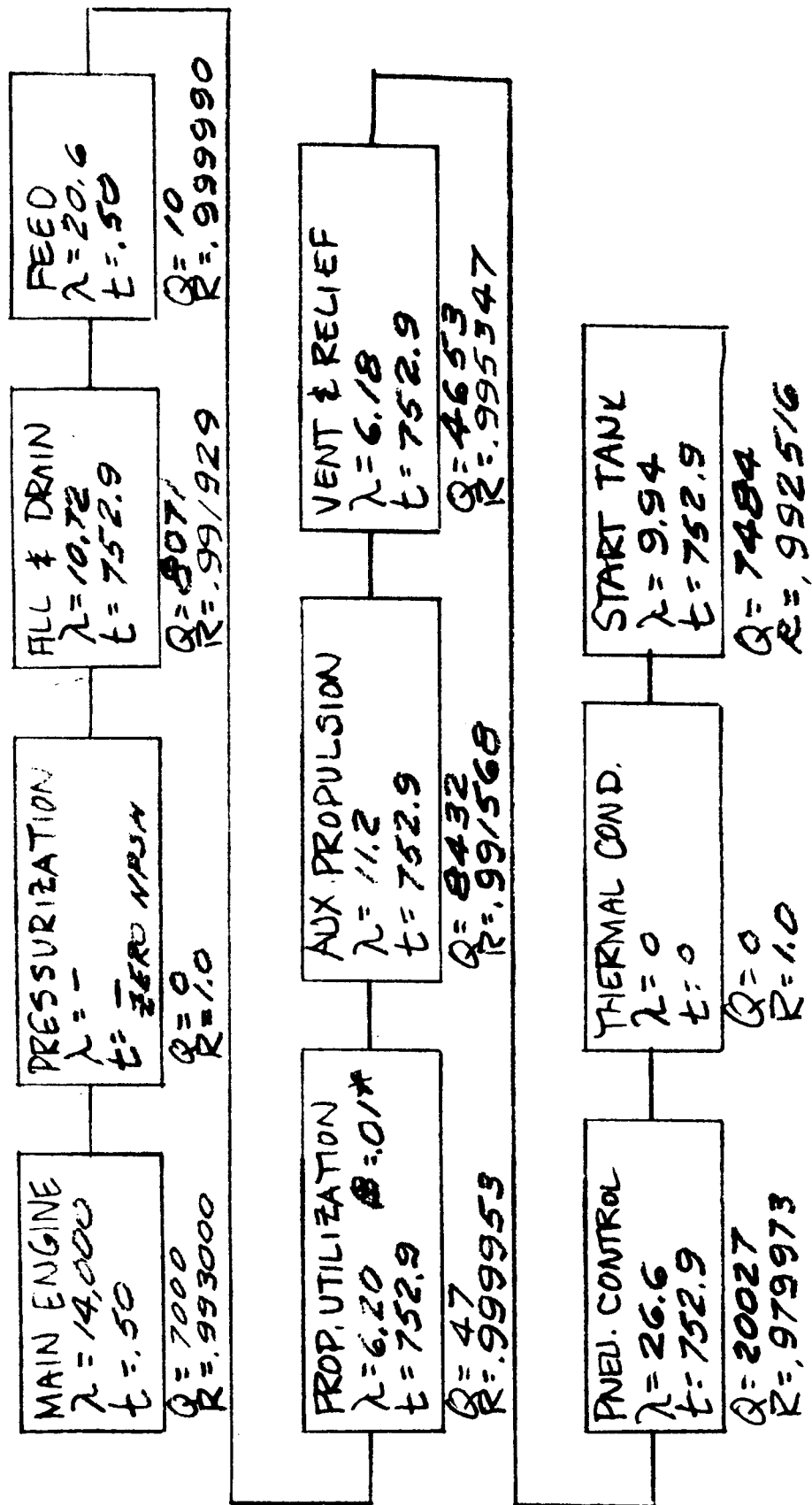


Figure 3-35 RELIABILITY PROJECTIONS - PROGRAM 403



NOTE: λ IN FAILURES $\times 10^{-6}$ HOURS

Figure 3-36 PROPULSION SUBSYSTEM RELIABILITY PREDICTION - PROGRAM 403

4.0 RECOMMENDED FUTURE EFFORT

A next level of detail that future activities could pursue would involve additional work in the areas of mission analysis, attitude control system considerations, thermal effects, and modes of astrionic subsystem operations. The following points present some of the activities and study areas that are recommended for further consideration.

1. Determine the velocity required to remain in lunar orbit and return as a function of lunar stay time, noting that the minimum energy case of 28,000 ft/sec occurs only for discrete stay times.
2. Perform additional extended synchronous mission analyses to include N satellites ($N > 4$) and examine the effect of angular orientation of the satellites visited.
3. Extend the loiter mission analysis to determine the effect on attitude control system (ACS) propellant expenditure of time varying vehicle mass due to propellant boiloff.
4. Analyze a more sophisticated control limit cycle to better establish the ACS propellants expended.
5. Perform analyses to determine whether the extended mission time can be adequately utilized to minimize the tug velocity expended for node adjustment for the tug interplanetary and return mission.
6. Define for the extended tug mission the thermal environment in more detail which the vehicle and vehicle tanks will encounter.
7. Examine the range of vehicle orientations in a given orbit from the thermal standpoint.

8. Conduct the feasibility of incorporation of the design which includes the aluminum shield over the LH_2 and LO_2 tanks with the vented hydrogen used to reduce LH_2 boiloff as well as other techniques for reducing the insulation and propellant weight penalty.
9. Define potential modes of operation for the tug astrionics and specific effects on astrionics equipment from other defined mission profiles.
10. Estimate heating requirements to maintain astrionics equipment above minimum allowable temperatures for maintenance of operating reliability.
11. Evaluate relative merits of providing required heat by leaving non-essential equipment turned on versus providing heating by separate resistance heating.
12. Perform a detailed mission extension assessment of specific program options detailed out of the study selection process.

Appendix A

SLUSH HYDROGEN CHARACTERISTICS

The following points summarize some of the desirable characteristics of slush or triple point hydrogen. Figures A and B graphically display these properties.

1. Slush hydrogen has greater density and more refrigeration capacity than liquid hydrogen.
2. Reduction in evaporation loss during storage.
3. Hydrogen slush reduces the quantity of hydrogen gas that boils off, because both the latent heat of fusion of the solid (25.02 Btu/lb) and the sensible heat (22.72 Btu/lb) of warming the liquid from the triple point (13.803°K) to the normal boiling points (20.278°K) must be absorbed before the hydrogen starts to boil at atmospheric pressure.
4. 191.6 Btu will raise the temperature of 5.44 lbs of 50% slush or 8.43 lbs of subcooled liquid hydrogen from the triple point to the boiling point with no evaporation loss.
5. Use of 50% slush at the triple point, 13.801°K, has the advantage of a density 15.6% higher than that of the liquid hydrogen at its boiling point.

6.
 - Liquid Hydrogen - At its boiling point, 20.268°K, has a density of 4.418 lb/cu.ft.
 - 50% slush hydrogen at the triple point, 13.801°K, has a density 15.6% higher.
 - Average density of 50% slush hydrogen is 5.107 lb/cu.ft.
7. Slush hydrogen allows combination of lower tankage weight, longer storage in space, lower insulation weight and lower weight of vented hydrogen.
8. The maximum solid content for a flowable mixture appears from experimentation to be just above 50% solid by weight.
9. Slush hydrogen is in equilibrium with vapor-solid-liquid only at the triple point of 13.8°K (24.8°R) and 52.8mm Hg (approximately 1.02 psia); therefore, the space vehicle tankage must be pressurized with helium above atmospheric pressure.
10. Advantage of 50% slush hydrogen over atmospheric pressure liquid hydrogen are due to its 36 Btu per lb higher heat capacity (lower enthalpy) and/or its 13% lower specific volume.

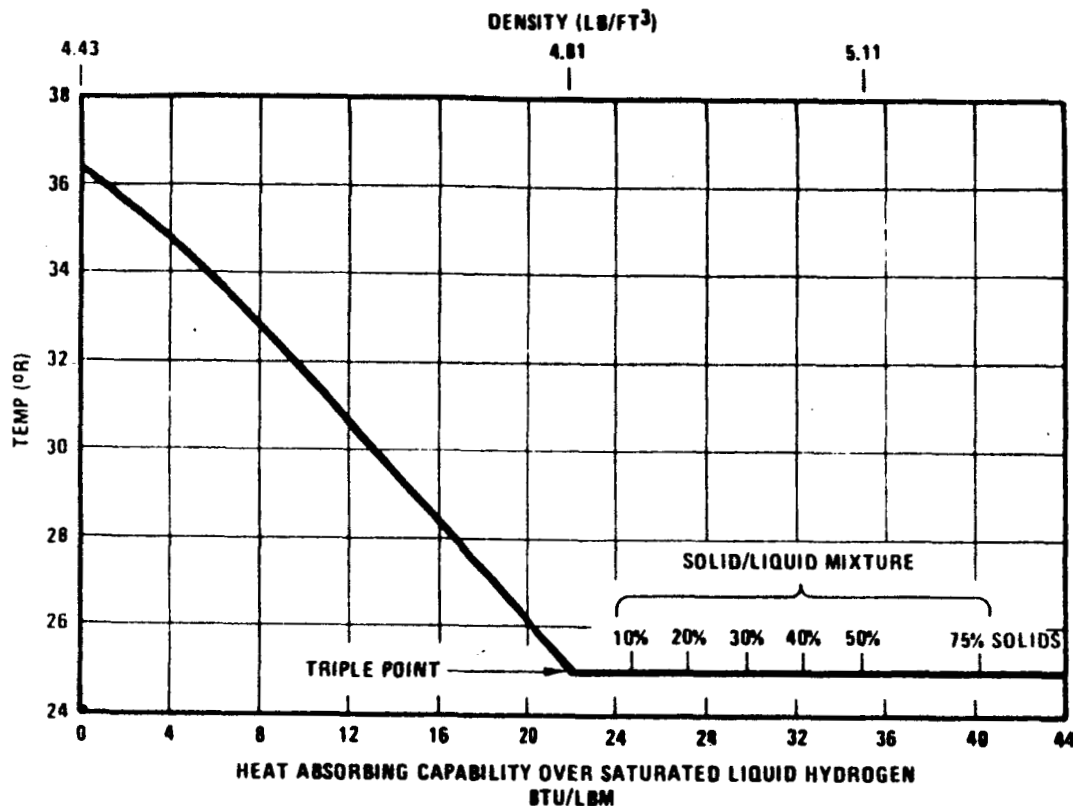


Figure A. SLUSH HYDROGEN CHARACTERISTICS

Depicted in this chart are the characteristics of saturated liquid hydrogen (at 1 atmosphere), sub-cooled liquid hydrogen and mixtures of solid hydrogen. The increased density and heat capacity of sub-cooled liquid and solid hydrogen mixtures can be advantageous when utilized as a space vehicle fuel. At zero Btu/lbm heat capability difference the indicated density of saturated liquid hydrogen is 4.43 lb/cu.ft. Triple point defines when vapor-solid-liquid occur. For sub-cooled liquid hydrogen the density is shown as 4.81 lb/cu.ft. At 50% solids, which tests has indicated can be flowed with no apparent difficulty, the density is 5.11 lb/cu.ft.

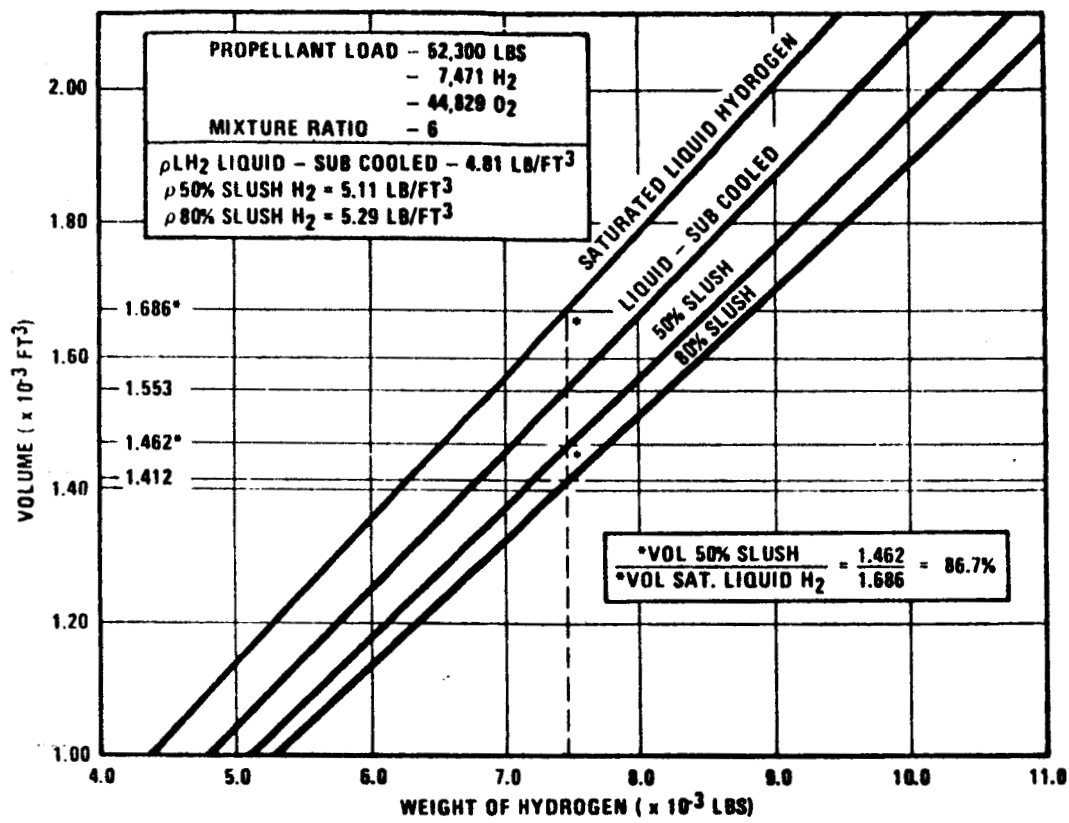


Figure B. VOLUME VS WEIGHT OF HYDROGEN FOR VARIOUS DENSITIES

Displayed graphically in this chart are the volumes and weights for saturated liquid hydrogen, sub-cooled liquid hydrogen, and 50% and 80% hydrogen slush for the range of values of interest for a Tug design. For a propellant load of 52,300 lbs and a MR of 6 7,471 lbs of hydrogen is required. As shown a 7,471 lb load of 50% hydrogen slush mixture would require a volume 13.3% less than that required for saturated liquid hydrogen.

The following calculations assume the thermal insulation is optimized for a 6-day mission.

Fifty percent hydrogen slush at melting point temperature = $434.4^{\circ}\text{F} = 25.6^{\circ}\text{R}$

$$\text{Heat of fusion} = 13.9 \text{ cal/gm} \times \frac{1.7988 \text{ BTU/lbm}}{1 \text{ cal/gm}} = 25. \frac{\text{BTU}}{\text{lbm}}$$

Heat of vaporization = $107 \text{ cal/gm} = 192.3 \text{ BTU/lbm}$

Hydrogen melting point temperature = $422.9^{\circ}\text{F} = 37.1^{\circ}\text{R}$

For 5.44 pounds of 50 percent slush: Hence this represents 2.72 pounds of solid hydrogen. Therefore the energy required for change of phase of 2.72 pounds of solid hydrogen at 25.6°R is:

$$Q_{S \rightarrow L} = m \cdot h_{\text{fusion}} = 2.72 \text{ lbm} \times 25 \frac{\text{BTU}}{\text{lbm}} = 68. \text{ BTU}$$

There remains now 5.44 lbm of liquid hydrogen at 25.6°R . The specific heat for the liquid hydrogen between the temperatures of 25.6°R to 37.1°R will be assumed to be:

$$C_p \cong C_s \cong \frac{1}{2} (2.4^{4.2} + 1.8) = 2.1 \text{ BTU/lbm} \cdot \text{R}$$

The energy required for raising the 5.44 lbm of liquid hydrogen from 25.6°R to 37.1°R is estimated as:

$$\begin{aligned} Q_2 &= m C_p \Delta T = 5.44 \text{ lbm} \cdot 2.1 \frac{\text{BTU}}{\text{lbm}^{\circ}\text{R}} (37.1^{11.5} - 25.6)^{\circ}\text{R} \\ &\cong 131.3 \text{ BTU} \end{aligned}$$

Therefore the absorbed energy required to raise the 5.44 lbm of 50 percent hydrogen slush from its 25.6°R to its saturation liquid temperature of 37.1°R is:

$$\begin{aligned} Q_{\text{TOT}} &= Q_{S \rightarrow L} + Q_2 = 68. + 131.3 \\ &= 199.3 \text{ BTU} \end{aligned}$$

Tug vehicle fuel capacity is 7471 lbm of 50 percent slush hydrogen. Therefore the required energy to raise 7476 lbm of 50 percent slush hydrogen from 25.6°R to saturated liquid hydrogen temperature of 37.1°R at one atmosphere pressure is:

$$Q_{\text{TOT TUG}} = 199.3 \text{ BTU} \times \frac{7471 \text{ lbm}}{5.44 \text{ lbm}} = 2.74 \times 10^5 \text{ BTU}$$

From Figure 3-20 the liquid hydrogen boiloff of 140 lbm propellant weight for a 6 day mission - continuous vent. Hence this represents a heat load for 6 day mission of:

$$\begin{aligned} Q_{6 \text{ day}} &= m_{\text{boiloff}} h_{fg} \quad 140 \text{ lbm} \times 192.3 \frac{\text{BTU}}{\text{lbm}} \\ \text{mission} &\approx 2.7 \times 10^4 \text{ BTU} \end{aligned}$$

This absorbed heat represents the tug design heat load for the 6 day mission.

CONCLUSION:

Based upon the, $Q_{6 \text{ day}}$, heat load for the six day mission, it can be said that no boiloff will occur when 50 percent slush hydrogen is utilized, since

$$\begin{array}{l} Q_{6 \text{ day}} < Q_{\text{TOT}} \\ \text{mission} \quad \text{TUG} \end{array}$$

For 50 percent slush hydrogen:

$$\text{Solid hydrogen mass} = \frac{1}{2} \times 7471 \text{ lbm} = 3735.5 \text{ lbm}$$

Energy required for change of phase of 3735.5 lbm of solid hydrogen at 25.6°R is:

$$Q_{S \rightarrow L} = m_i h_{\text{fusion}} = 3735.5 \text{ lbm} \times 25 \frac{\text{BTU}}{\text{lbm}} \approx 9.33 \times 10^4 \text{ BTU}$$

$$\text{NOTE THIS HEAT FLUX GREATER THAN } Q_{6 \text{ day}} = 2.7 \times 10^4 \text{ BTU.}$$

mission

Hence with 50 percent slush hydrogen, 3735.5 lbm, and with the 6 day mission heat load of 2.7×10^4 BTU, the following represents the mass of solid hydrogen that would be melted at this heat load

$$\begin{aligned} m_6 &= \frac{Q_{6 \text{ day}}}{h_{\text{fusion}}} = \frac{27 \times 10^3}{25 \text{ BTU/lbm}} = 1.08 \times 10^3 \text{ lbm} \\ &= 1080. \text{ lbm} \end{aligned}$$

Mass of solid hydrogen remaining after 6 day mission:

$$m_{\text{solid}}]_{6 \text{ day}} = m_i - m_6 = 3735.5 - 1080. = 2655.5 \text{ lbm}$$

Hence at end of 6 day mission the remaining propellant slush consistency would be:

$$\text{Slush Concentration Remaining} \approx \frac{2655.5}{7471.} = 35.5\%$$

Hence a slush consistency of $\frac{1080 \text{ lbm}}{7471 \text{ lbm}} \approx 14.45\%$ with the referenced 6 day heat load of 2.7×10^4 BTU would leave a fully liquified hydrogen propellant at the temperature of 25.6°R at the end of:

$$\frac{6 \times 60}{14.45} = 20.6 \text{ days}$$

Now in the event, it is required to have the liquid hydrogen total mass, 7476 lbm, to be at the saturation temperature of 37.1°R ; the following calculations will report the initial slush consistency to achieve this.

The energy required to raise 7476 lbm liquid hydrogen from 25.6°R to 37.1°R :

$$\begin{aligned} Q_{L \rightarrow} &= mC_p \Delta T = 7471 \text{ lbm } 2.1 \frac{\text{BTU}}{\text{lbm}^\circ\text{R}} \times 11.5^\circ\text{R} \\ &= 1.807 \times 10^5 \text{ BTU} \end{aligned}$$

This is greater than $Q_{6 \text{ day mission}}$, hence slush not required. But a subcooled liquid may be utilized. The degree of subcool is estimated as follows:

$$\begin{aligned} \Delta T &= \frac{Q_{6 \text{ day mission}}}{mC_p} = \frac{2.7 \times 10^4 \text{ BTU}}{7471 \text{ lbm } 2.2 \frac{\text{BTU}}{\text{lbm}^\circ\text{R}}} \\ &= 1.643^\circ\text{F} \end{aligned}$$

The time required to heat the entire mass to the saturation temperature is $\frac{1.807 \times 10^5}{2.7 \times 10^4} \times 6 = 40.2$ days. The total time from a 50 percent slush to saturated liquid is then $20.6 + 40.2 = 60.8$ days.